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Orbiter External Hydrogen Tank Study

Volume I. Summary

Contract NAS9-10960
DRL T-678, DRL Line Item 1
DRD SE-382T
SD 71-141-1
25 June 1971

Space Division • North American Rockwell   12214 Lakewood Boulevard, Downey, California 90241
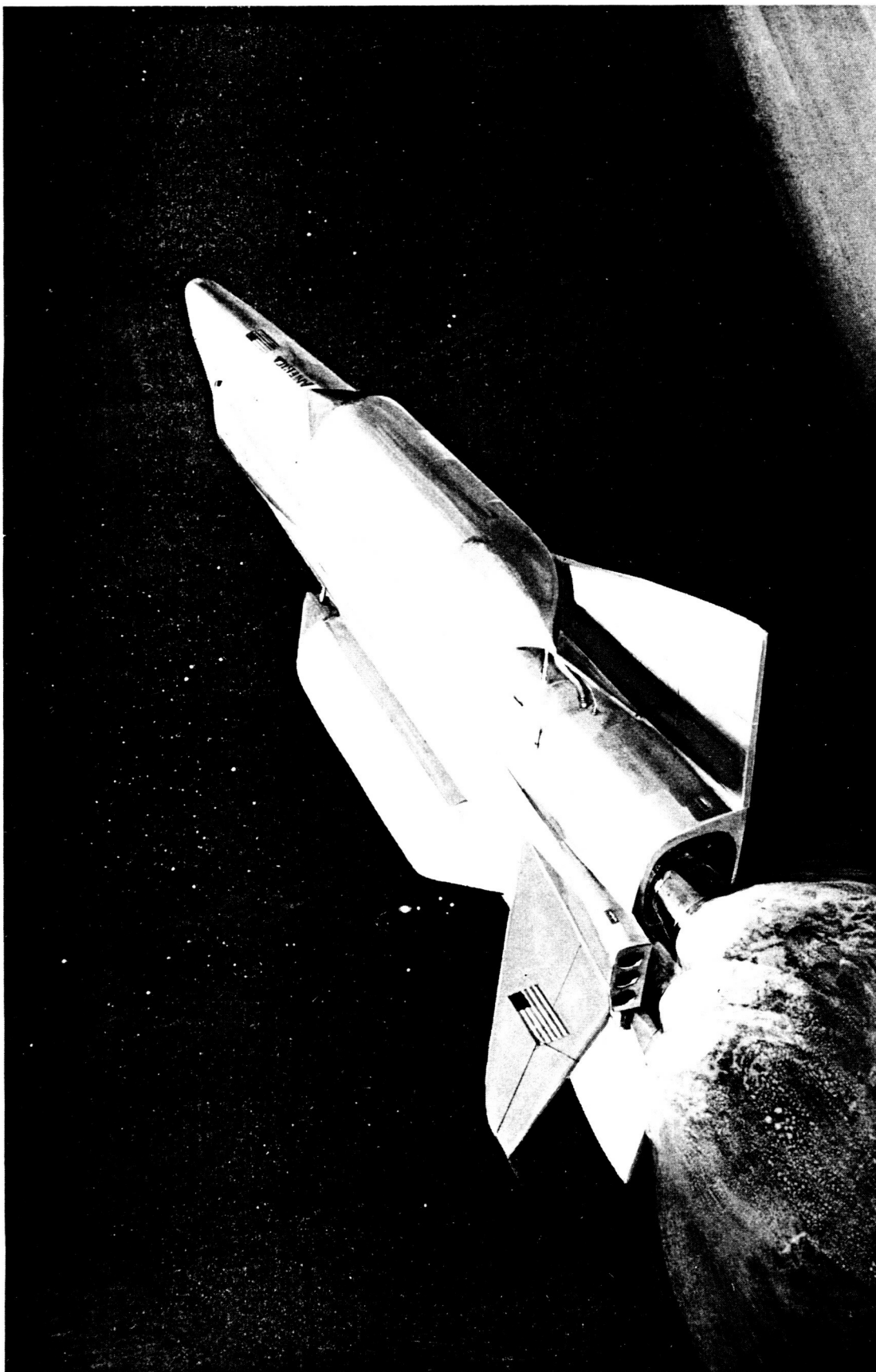
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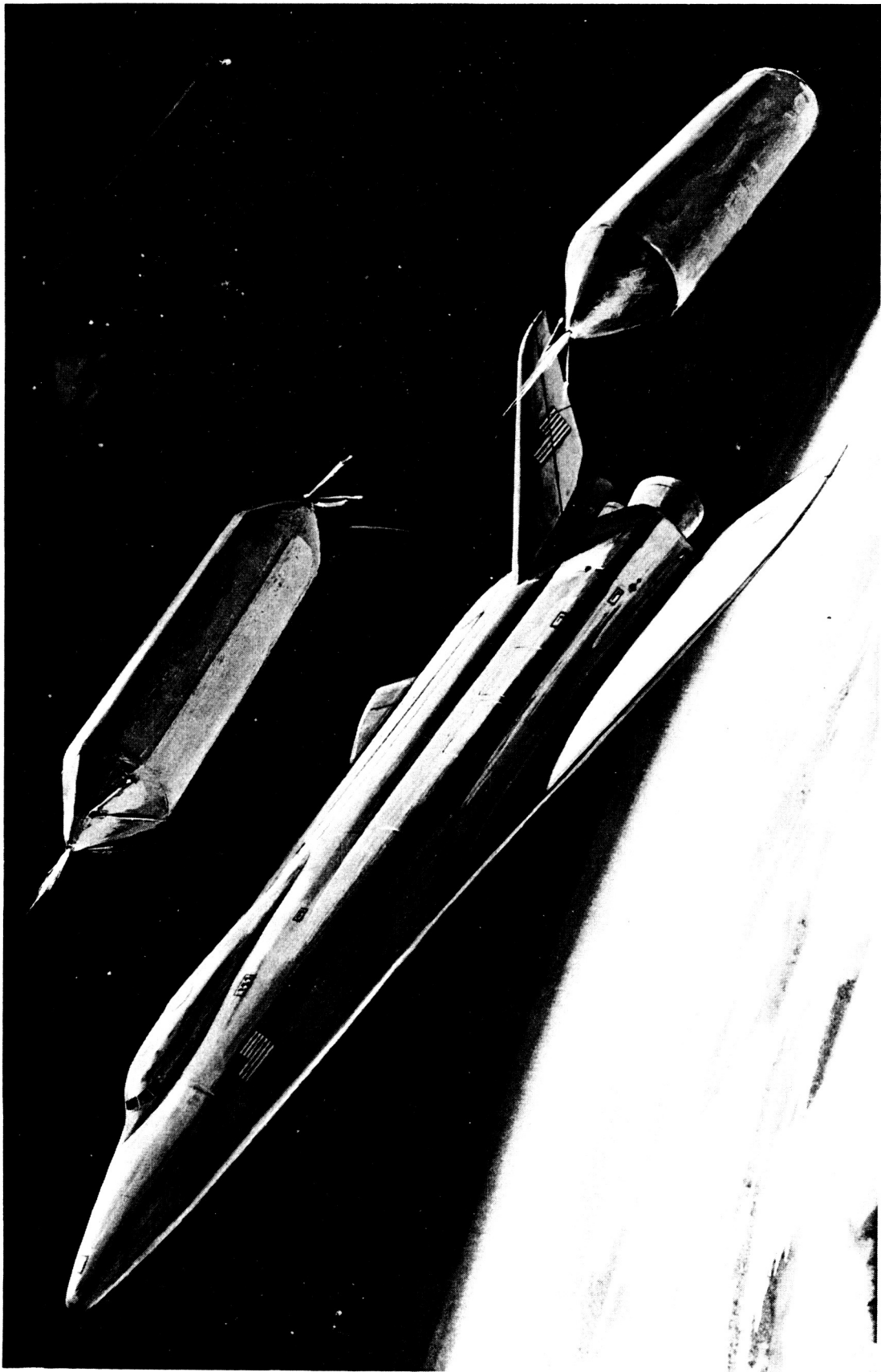# ORBITER EXTERNAL HYDROGEN TANK STUDY

Volume I
Summary

Contract NAS9-10960,
DRL T-687, DRL Line Item 1
DRD SE-382T

Approved by

B. Hello
Vice President and General Manager
Space Shuttle Program

FOREWORD

This report is submitted to NASA in accordance with Contract NAS9-10960. The report documents the results of the Study of a Space Shuttle Vehicle Using Orbiter External Hydrogen Tanks and a Heat Sink Booster. The study was performed as an element of the Phase B Space Shuttle Definition Program under the direction of the Space Division of North American Rockwell, Downey, California. Other members of the team were: Convair Aerospace Division of General Dynamics and Aerospace Division of Honeywell, Inc.

This document is Volume 1 of the study final report. It presents a summary design and program definition of the space shuttle with orbiter external hydrogen tanks. The concept is also compared with the fully reusable space shuttle system.

Volumes of the report for the study of the orbiter external hydrogen tank concept are as follows:

Volume 1 - Summary of the Study of Space Shuttle With Orbiter External Hydrogen Tanks

Volume 2 - Configuration Description of the Space Shuttle with Orbiter External Hydrogen Tanks (Book 1 and Book 2)

        Appendix A Drawings

        Appendix B Structural Analysis

Volume 3 - Programmatic Impact of the Space Shuttle With Orbiter External Hydrogen Tanks

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INTRODUCTION

1.0 INTRODUCTION

During 1970 and 1971, NASA directed Phase B studies of space shuttle systems which offered potential for reduction in cost of delivering payloads into low earth orbit. The economic advantages of these shuttle systems are achieved through multiple reuse of the vehicle as opposed to using expendable launch vehicles for a single mission.

Studies directed by North American Rockwell encompassed (1) two-stage reusable shuttle systems and (2) shuttle vehicles using orbiters with external hydrogen tanks and booster stages in which the structure and propellant tankage are used as a heat sink.

This report summarizes the results of analyses of the design, performance, and program implications of the latter concept. The concept offers potential for reducing development cost compared to a fully reusable system. The development cost reduction is made possible by the reduced size of the vehicle required when orbiter hydrogen tanks are jettisoned in orbit.

Studies of this shuttle system were initiated in April 1971 and terminated on June 30, 1971. The studies were directed toward definition of the design to accomplish specific missions and also identification of changes to the reusable shuttle program plans including revised test, manufacturing, and operational requirements. To assess the merit of this system, it also was compared to a fully reusable space shuttle vehicle.

The study was performed in two phases as illustrated by the schedule presented in Figure 1-1. Activity during the first six-week period was directed toward evaluation of a number of vehicle configurations to select one system for preliminary design analysis during the latter phase of the study.

During the study, emphasis was placed on the analysis of design and operational features of the external hydrogen tank system which were unique to the concept and therefore influenced its relative merit compared to the fully reusable vehicle.

First-phase activities encompassed vehicle sizing, conceptual design definition, and estimation of program costs for a number of systems using orbiter external hydrogen tanks. Systems with two-engine and three-engine orbiters using external hydrogen tanks and various main LO_2 tank arrangements were configured. The studies showed that the concept is feasible and
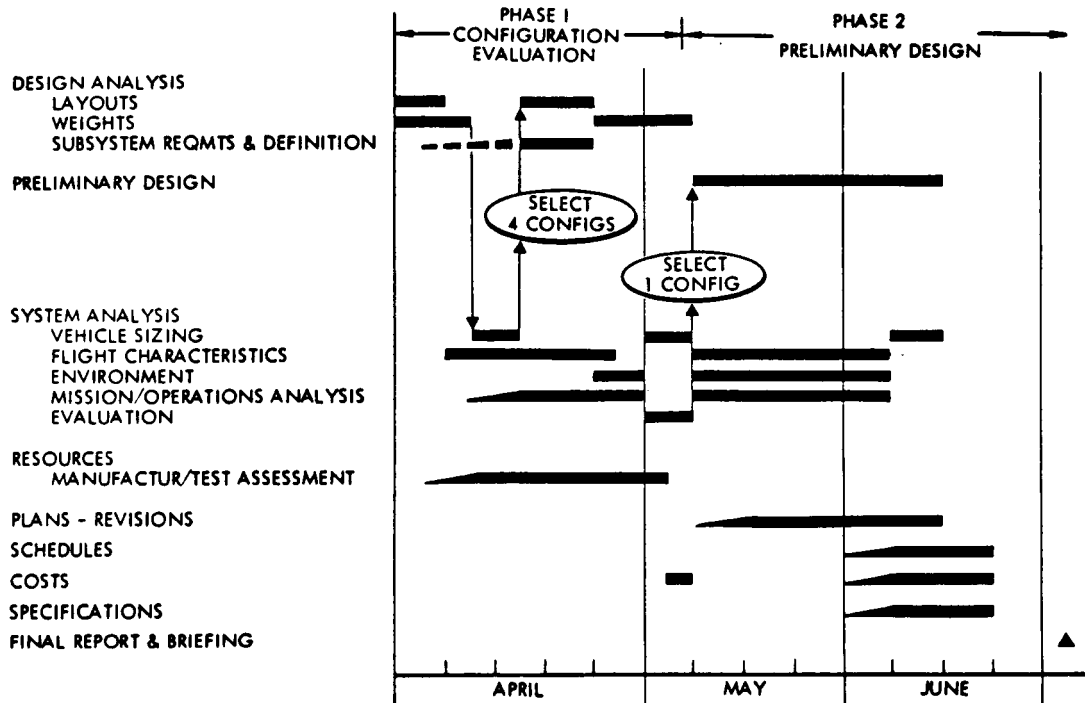
Figure 1-1. Approach and Schedule

that tank disposal is safe. The two-engine orbiter required high staging velocity which necessitated a heat shield on the booster. A system with a three-engine orbiter, however, reduced vehicle size and staging velocity and facilitated use of a heat sink booster. This system also minimized program cost.

Fully reusable vehicles with two-engine and three-engine orbiters were also synthesized and compared with the expendable tank concept. The fully reusable system with three-engine orbiter was approximately 400,000 pounds heavier at lift-off than the expendable tank system and also required greater expenditures for development. Operational costs, however, were lower and over a ten-year program, total expenditures were less than for the orbiter external hydrogen tank concept.

The second-phase activities concentrated on preparing a more in-depth design definition of the selected three-engine orbiter external hydrogen tank (OEHT) concept, evaluation of resource requirements, and preparation of new cost estimates. The flight characteristics and design environment were established for each vehicle element, booster, orbiter, and external tanks. Based on this information, a preliminary design was prepared for the heat-sink booster and three-engine orbiter including a detail definition of the

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expendable tanks. Further analysis was conducted which verified the
the acceptablility of tank disposal even when tank breakup on entry is
allowed. Using the preliminary design definition, the programmatic plans
and detail cost estimates were prepared.

To facilitate comparison of the fully reusable shuttle system with the
·OEHT, the definition of the reusable shuttle system with a three-engine
orbiter was updated.

The reusable system GLOW is approximately 580,000 pounds greater
than the OEHT system. The OEHT system development cost is less than
that for the fully reusable; however, because of the recurring expendable
tank costs, the total program cost over a ten-year period is slightly greater.
The cost crosssover occurs after approximately 350 flights.

2.0 MISSION AND SYSTEM REQUIREMENTS

This section of the report provides a summary of the missions and system requirements which have the most significant impact on vehicle design. Table 2-1 identifies the requirements and their influence on the space shuttle vehicle system.

Table 2-1. Orbiter External Hydrogen Tank Shuttle System
Program Requirements Summary

Requirements	Influence/Effect on Vehicle System
Reusable two-stage system Orbiter external LH_2 tanks Vertical takeoff - horizontal landing	o Parallel arrangement for ascent to minimize loads, control requirements o Disposable tanks to reduce development cost o Shape orbiter and booster to achieve good subsonic aerodynamic landing characteristics
Cargo bay 15 ft diameter by 60 ft long	Orbiter volume, weight, and total vehicle system weight

Missions
  65,000-lb payload
    Orbit: 28.5-deg inclination,
        100 nm
  On orbit ΔV, 900 fps
 	
  25,000-lb payload
    Orbit: 55-deg inclination,
        270 nm
  On orbit ΔV, 1500 fps
 	
  40,000-lb payload
    Orbit: 90-deg inclination,
        100 nm
  Orbit ΔV, 650 fps
	Vehicle system size and propellant distribution

Table 2-1. Orbiter External Hydrogen Tank Shuttle System
Program Requirements Summary (Cont)

Requirements	Influence/Effect on Vehicle System
Orbiter expendable LH$_2$ tanks	Orbiter internal design arrangement and structure Propellant and electrical systems Vehicle size Orbital operations

Orbiter hypersonic crossrange requirement
    1200 nm nominal
    (return to launch site)
	High hypersonic L/D orbiter aerodynamic shape, thermal protection system, orbiter weight, orbiter guidance and control requirements, ACPS requirements, and entry mode selection.

1500 fps on orbit ΔV capability, tanks sized for 2000 fps for design mission	Orbiter on-orbit propellant tankage, weight, vehicle system weight, and abort capability

Go-around capability for booster and orbiter
    Booster flyback to base
    Orbiter and booster ferry
        capability
    JP fuel for ABES
    Commonality
	Vehicle size and weight, ABES engine selection, ascent, trajectory parameters, and orbiter and booster vehicle arrangement

Operation between zero and maximum payload Operation without ABES	Orbiter vehicle arrangement for c.g. control and aerodynamic controls design
Mission duration, 7 days	Subsystem design, TCS design, and vehicle weight and size

Shirtsleeve environment for crew
    and passengers
Environment compatibility with
    space station
	ECLSS design, pressure vessel compartment, and passenger module selection

Table 2-1. Orbiter External Hydrogen Tank Shuttle System
Program Requirements Summary (Cont)

Requirements	Influence/Effect on Vehicle System
Safe mission termination capability, safe egress, and intact abort Once-around return capability with one orbiter engine out at or after separation	Vehicle sizing, OMS engine selection, ascent trajectory parameters, and booster flyback requirements; hatchways and access provisions
LO_2-LH_2 propellants, main engine ICD except F_{SL} = TBD	Vehicle sizing, selection of the number of engines on booster and orbiter
FO/FS for subsystems except structure and pressure vessels	Subsystem concept selection, weight and cost, and vehicle weight and size
Maximum axial load factor equal 3 g's	Vehicle weight and size and performance

PHASE 1

3.0 CONFIGURATION EVALUATION

This section summarizes results of the first phase of the study of the space shuttle with orbiter external hydrogen tanks (OEHT). This phase of the program was directed toward definition of design requirements and evaluation of a number of orbiter and external hydrogen tanks designs to select the best configuration for preliminary design analysis during Phase 2 of the study. Investigations also were performed on boosters using (1) radiative heat shield and (2) heat sink for thermal control to assess the relative merits of these concepts. Program requirements and cost for the OEHT candidate systems were established. Reusable vehicle configurations and costs were also defines and compared with the OEHT system.

Vehicles using orbiter external hydrogen tanks were configured to accomplish the missions and satisfy the program requirements summarized in Section 2.0.

The orbiter and booster for the OEHT system maintains the arrangement used by the fully reusable space shuttle with the orbiter mounted on the upper surface of the booster. Subsystem concepts of the reusable vehicle are also maintained in the OEHT system unless a unique design or operational requirement prevents their retention.

Key issues which were addressed in the study included (1) determination of system requirements for safe disposal of orbiter LH_2 tanks, (2) vehicle performance, and (3) program costs. A summary of key issues is presented in Table 3-1 with comments on the assessed merit of the OEHT concept based on Phase 1 study activities.

3.1 VEHICLE SIZING

During the first phase of the study, the sizes and desired main engine thrust levels for a number of OEHT vehicles were computed to satisfy mission requirements. Candidate vehicle types are illustrated in Figure 3-1. As illustrated, the orbiters encompass a number of different main propulsion systems and main LO_2 tank arrangements and have the external main hydrogen tanks mounted on the fuselage. Boosters include designs with radiative heat shields and using structure as a heat sink.

Table 3-1. Key Issues

Issue	Status	Comment
Is OEHT concept capable of performing missions?	No deficiency identified	• Acceptablity of heat sink Booster sensitive to orbiter mass fraction/staging conditions • High ΔV missions limited by heat sink booster
Is tank disposal acceptable?	Shipping or land mass impact probability extremely low	• Simple tank design with TPS for ascent
Is orbiter design acceptable?	Concept is feasible and reusable vehicle sub-systems are applicable	• Wind tunnel data required
Is heat sink booster acceptable?	Feasible for computer staging conditions	• Weight lower than TPS booster
Is OEHT weight lower than 2-stage reusable?	GLOW and combined dry weight lower 3-engine orbiter	• Weight difference between OEHT and reusable vehicle minimized for 3 engine orbiters
Is OEHT DDT&E lower than 2-stage reusable?	DDT&E cost appears lower	
Is OEHT operational cost lower than 2-stage reusable?	• Cost of expendable tank increases the OPS costs over the reusable vehicle costs • Program cost higher for OEHT concept	• Detailed cost analysis is required as merit of system dependent on tank cost and booster maintenance cost • Program cost sensitive to tank cost estimate
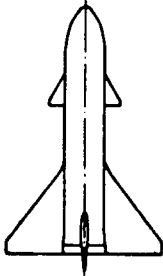
CONFIG NO.	ORBITER		VEHICLE		BOOSTER	
	INTERNAL TANK LOCATION	NO. ENGINES	ENG THRUST	ABORT	AL	TPS
1	LO2 FWD (LOX TANK, EXTERNAL TANKS, CARGO BAY)	2	550	YES		X
2		3	550	YES	X	
3		2	VAR	YES	X	
4		3	VAR	YES	X	
5		2	550	NO	X	
6	LO2 MID (LOX TANKS, EXTERNAL TANKS, CARGO BAY)	2	550	YES		X
7		3	550	YES	X	

RADIANT HEAT SHIELD OR HEAT SINK

Figure 3-1.   Vehicle Configuration Options

Orbiter configurations include high fineness ratio bodies using forward mounted main LO2 tank and vehicles with low fineness ratio bodies with main LO2 tanks mounted in the mid-body section.   Vehicles analyzed included two- and three-engine orbiter systems in which the optimum thrust to minimize system cost was established.   Vehicles using two engines with 550,000-lb sea level rated thrust on the orbiter require a high staging velocity to facilitate abort to a once around orbit following one main-engine failure. This high staging velocity necessitates use of a heat shield on the booster. Where three engines are used on the orbiter or where the thrust level is increased, the velocity extracted from the orbiter may be increased and therefore allows a low staging velocity and use of a heat sink booster.

The sizes of the vehicles computed during the first phase of the study are illustrated in Figure 3-2.   The figures illustrates gross lift-off weight, combined dry weight of the orbiter and booster, and staging velocity as a function of engine thrust and number of orbiter engines.

In sizing the vehicle to deliver a 40,000-lb payload into a 100-nm polar orbit, the two-engine orbiter results in a vehicle gross lift-off weight between 4.5 and 4.7 million pounds.   The dry weight variation is negligible for a main engine thrust range from 450 to 650 pounds.   Detail information was available for an engine delivering a 550,000-lb sea-level thrust, and

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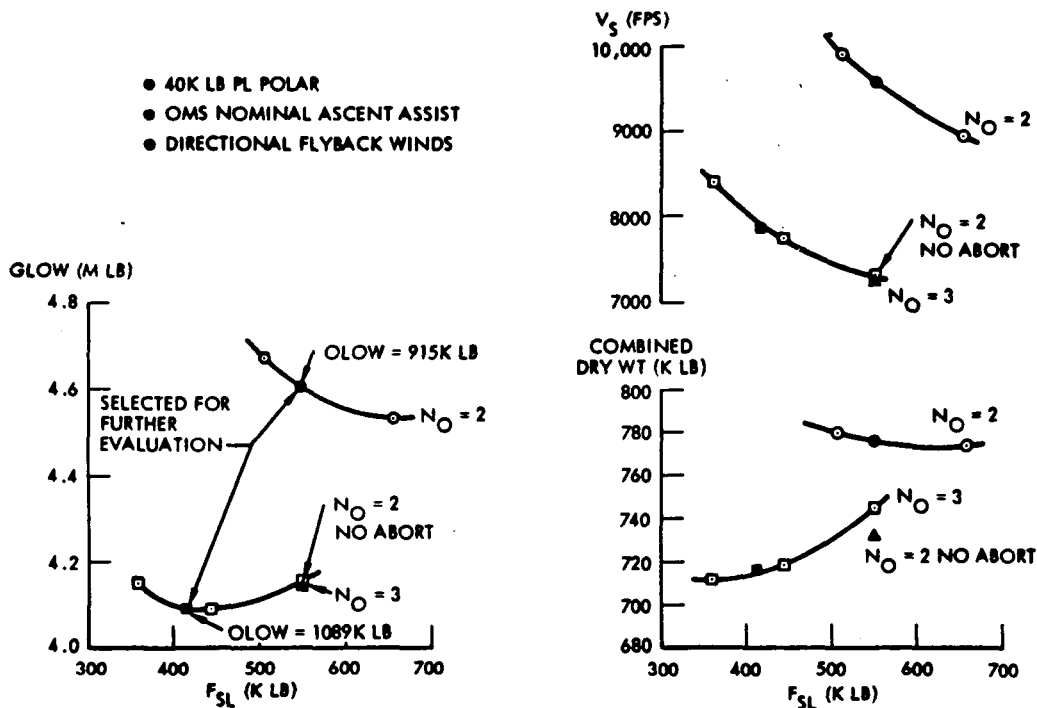
Figure 3-2. Vehicle Sizes

therefore, a system using two engines at this thrust level was selected for further analysis. This system provides a staging velocity of approximately 9600 fps and requires a radiative heat shield on the booster. A vehicle configured to satisfy the previously described mission and using three engines on the orbiter results in a gross lift-off weight of approximately 4.1 million pounds with the minimum weight being achieved with engine sea-level thrust of 415,000 pounds. This vehicle provides a staging velocity of approximately 7800 fps and facilitates use of a heat sink booster. It is noted that the critical factor in defining the vehicle size is the requirement to provide capability for abort into a once-around orbit following a main-engine failure on the orbiter. The specified polar mission requires 650 fps for on-orbit maneuvers. Mission abort following a main-engine failure on the orbiter is achieved through providing additional propellant in the orbit maneuvering tank. This propellant is burned through the orbit maneuvering system in parallel with the main engines to provide abort capability. The penalty in providing abort capability is greater in a two-engine orbiter than in a system using a three-engine orbiter. This is due to the greater percentage of thrust loss associated with this configuration. For the polar mission, therefore, with the 650 fps required for on orbit maneuvers, a three-engine orbiter results in a lower gross lift-off weight and a lower combined booster and orbiter dry weight. As illustrated by the figure, the lift-off weight reduction is approximately 500,000 pounds. A similar weight reduction can be achieved through elimination of the once-around abort requirement in a vehicle using a two-engine orbiter. Sizes and flight characteristics of the vehicles identified in Figure 3-1 are summarized in Figure 3-3.

CHARACTERISTICS	CONFIGURATIONS						
	1	2	3	4	5	6	7
ORBITER							
ENGINES	2	3	2	3	2	2	3
THRUST K LB	550	550	650	415	550	550	550
OLOW K LB (INCLUDES TANKS)	915	1200	1000	1088	1200	915	1200
DRY WT K LB (W/O TANKS)	192	222	206	206	210	192	222
ABORT	✓	✓	✓	✓	X	✓	✓
FWD LOX TANK							
2 CELL	✓	✓	✓	✓	✓	✓	✓
2 TANKS				✓			
SINGLE TANK				✓			
EXTERNAL LH$_2$ TANK(S) DRY WT	22.3	29.9	24.5	27.1	29.2	22.3	29.2
BOOSTER							
ENGINES	11	10	9	13	10	11	10
THRUST K LB	550	550	650	415	550	550	550
BLOW K LB	3690	2955	3532	3004	2953	3690	2955
DRY WT K LB	557	493	543	483	493	557	493
MAJ HEAT SINK				✓			
TPS	✓						
V$_S$ (K FPS)	9.6	7.3	8.9	7.8	7.3	9.6	7.3
GLOW (M LB)	4.6	4.2	4.5	4.1	4.2	4.6	4.2
SELECTED CONFIGURATIONS	✓			✓			

Figure 3-3. Candidate Vehicle Summary

3.2 INTEGRATED VEHICLE DEFINITIONS

Based on the previously described sizing, two vehicle types were selected for conceptual design analysis. These vehicles are defined in Table 3-2 with associated variations in orbiter external hydrogen tank configuration and arrangements in the main LO_2 tank. The mission profile for the two vehicle categories is presented in Figure 3-4.

The integrated vehicle assembly using two-engine orbiter is presented in Figure 3-5. As shown, the orbiter is mounted forward on the upper surface of the booster to maintain adequate clearance during separation. A similar separation system to that used on the fully reusable baseline system is provided. As shown, the vehicle lift-off weight is 4.7 million pounds when the propellant is loaded to satisfy the 40,000-lb payload mission.

The integrated vehicle using three-engine orbiters is illustrated in Figure 3-6. These vehicle assemblies maintain the same orbiter/booster arrangement as the fully reusable system. Three orbiter configurations are identified:

1. System using a two-cell main LO_2 tank located forward

2. System with two separate main LO_2 tanks located forward

3. System with single main LO_2 tank located forward

Each of the vehicles illustrated utilizes cylindrical external hydrogen tanks. Weights for the three configurations are comparable and provide a lift-off weight of approximately 4 million pounds.

Table 3-2. First Phase Selected Options

Configurations		Variations		
No. 1 Orbiter	2 x 550,000-lb engines 2-cell LO_2 tank Shape - similar to baseline			
Tanks	Cylindrical No orientation Intact to 200,000 ft	• Cylindrical tank with pre-deorbit orientation • Tapered tank + aero orientation • Splayed tank • Cylindrical tank with breakup allowed		
Booster	11 x 550,000-lb engines RAD H/S			
No. 2 Orbiter	3 x 415,000-lb engines 2-cell LO_2 Shape similar to baseline			
Tanks	Cylindrical No orientation		TPS • No TPS • Outside SOFI • Inside SOFI/ ablator • Outside SOFI/ ablator	Separation • Links • Thrusters • Springs • Pistons • Manipulator
Booster	13 x 415,000-lb engines Heat sink			
No. 3 as No. 2	But orbiter with 2 separate LO_2 tanks			
No. 4 as No. 2	But orbiter with single single LO_2 tank			

TANK SEPARATION & DEORBIT

CIRCULARIZATION
100 X 100 N MI

ORBIT INJECTION
50 X 100 N MI

- **BOOSTER/ORBITER STAGING**

	2 ENG ORB SYS	3 ENG ORB SYS
V (FPS) =	9922	7719
h (FT) =	261K	207K
γ (DEG) =	10	14
q, (PSF) =	2	13.1

- MAX. q (PSF)
  469,508

- T/W_o
  1.283, 1.347

	• 2 ENG ORB SYS	• 3 ENG ORB SYS
R_c (N MI)	432	241

ETR

TANK IMPACT TARGET INDIAN OCEAN
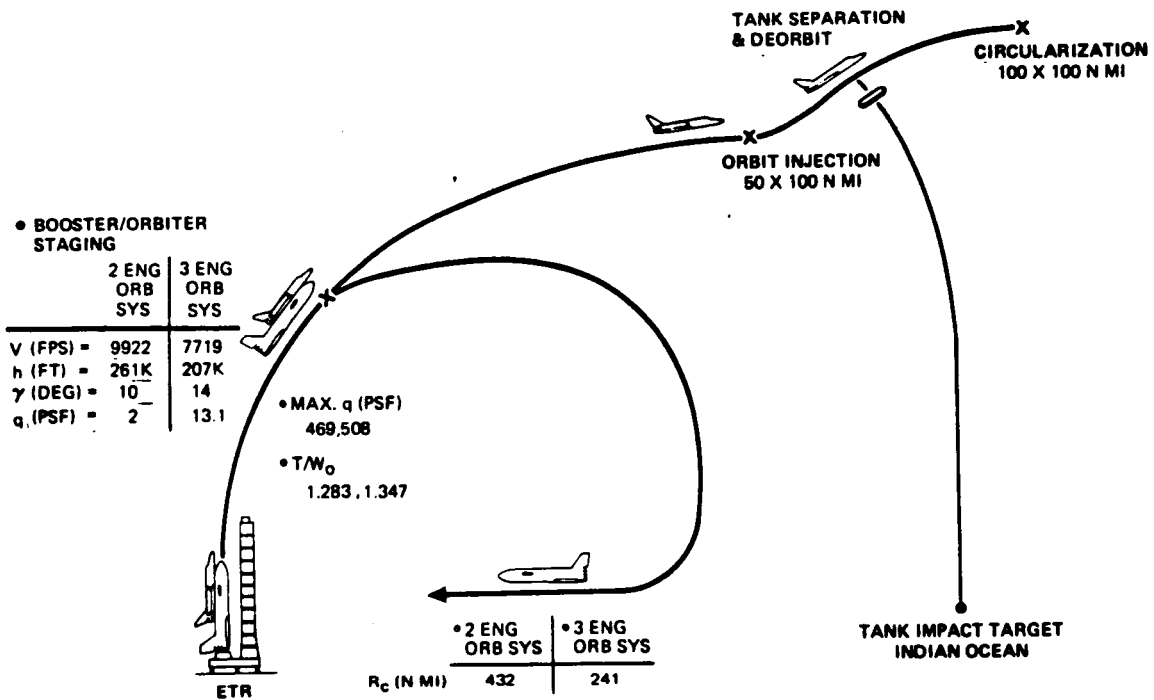
Figure 3-4. Mission Profile, Polar Orbit

ORBITER LOCATION ON BOOSTER COMPATIBLE WITH BASELINE
- $\gtrsim$ ORBITER ENGINE TO BOOSTER RUDDER CLEARANCE
- ~ ORBITER TO BOOSTER NOSE OVERHANG
- SAME LENGTH DISPLACEMENT LINKS
- UTILIZE BOOSTER INTERSTAGE FRAME
- WITHIN BOOSTER TVC EXCURSION ± 10°

WEIGHTS (LB) (40,000 LB PAYLOAD)	
LIFTOFF	4,716,223
ORBITER	783,965
EXTERNAL TANKAGE	105,026
BOOSTER	3,827,232

STA 950
STA 1750
49 5 FT (45 FT BASELINE)

0020 ORBITER (2) CELL LOX TANK

3007 EXTERNAL TANKAGE

(2) 632K VAC ENGINES

B-9W BOOSTER

(11) 550K SL ENGINES

101.0 FT

6 FT (20.5 FT BASELINE)

GIMBAL PLANE

234.0 FT
264 FT

134.6 FT

STA 1908
STA 2678
STA 3808
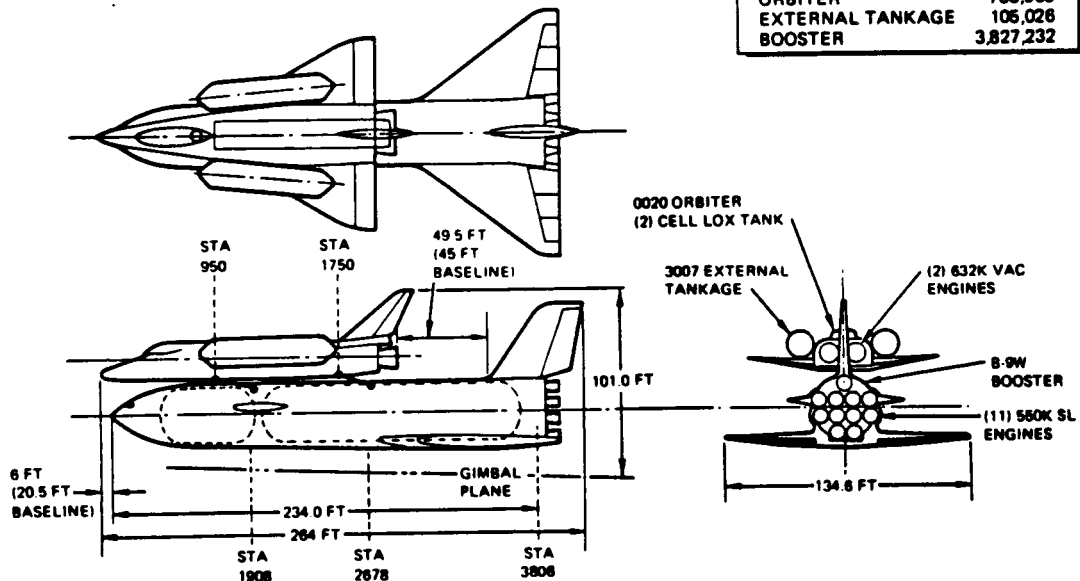
Figure 3-5. Integrated Vehicle Assembly,
Two Engine Orbiter

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Figure 3-6. Integrated Vehicle Assembly, Three Engine Orbiters

3.3 CANDIDATE ORBITER CONCEPT DESCRIPTIONS

This section describes the two- and three-engine orbiters incorporated in the integrated vehicles selected for conceptual design analysis.

3.3.1 TWO-ENGINE ORBITER

The two-engine orbiter selected for conceptual design analysis is illustrated in Figure 3-7. Major revisions to the baseline reusable vehicle include relocation of the hydrogen from the nose of the baseline system into two cylindrical tanks mounted on the side of the orbiter fuselage.

The main LO_2 tank is an integral two-cell tank arrangement located in the nose of the vehicle. The revised tank arrangement and required propellant load to satisfy mission requirements result in an orbiter body length which is approximately 370 inches less than that for the fully reusable orbiter. The vehicle including external LH_2 tanks is approximately 889,000 pounds at lift-off. The basic structure concepts are the same as those used in the fully reusable system but the main LO_2 tank becomes an integral load-carrying tank as opposed to the floating tank design in the fully reusable vehicle. Thermal protection is also achieved through the use of a reusable external insulation. This insulation system requires an increase in area compared to the fully reusable vehicle to account for ascent interference heating effects between the external hydrogen tanks and the body and wing of the orbiter. Propulsion system concepts are identical to those used in the fully reusable system and are defined in Figure 3-7.

The subject orbiter has a shape similar to that of the reusable vehicle and results in comparable aerodynamic characteristics. The maximum hypersonic L/D is approximately 2.2, and when entering at a 30-degree angle of attack, the vehicle has a hypersonic L/D of 1.6 which is identical to that of the two-engine fully reusable orbiter. In order to achieve trim capability, the wing is moved forward 50 inches compared to the baseline system to account for approximately one-percent movement of the c.g. Subsonic aerodynamic characteristics of the orbiter with external hydrogen tanks are also comparable to the fully reusable system. A slight increase in maximum subsonic L/D to a value of 7.9 is achieved through a higher ratio of wing area to planform area in order to maintain similar touchdown speeds.

The external hydrogen tanks used with this orbiter are of an aluminum monocoque construction with internal foam insulation and an external ablator to maintain an intact tank on reentry. (This tank design was revised during

Figure 3-7. General Arrangement, Two-Engine 0020 Orbiter

CHARACTERISTICS	
GROSS WT	• 888,991 LB (INCL EXTERNAL TANKS)
PAYLOAD	• 40,000 LB
AERO/ THERMAL	• ENTRY L/D = 1.61 • SUB L/D = 7.9 • W/C_L S = 85
PROPULSION SYSTEMS	• MAIN ENGINES = (2) 632K VAC • ACPS/OMS INTEGRATED TANKAGE • 29 ACPS THRUSTERS AT 2100 LB EACH • 3 OMS THRUSTERS AT 10,000 LB EACH • 4 JTF22A-4 ABES WITH JP FUEL
ORBITER STRUCTURE/ TPS	• LO_2 TANK-2-CELL INTEG ALUM • PRIMARY - TITANIUM • HEAT SHIELD - REI
EXTERNAL LH_2 TANK	• ALUM MONOCOQUE • INTERNAL FOAM INSULATION • ABLATOR • SEP SYSTEM PISTONS • SPIN MOTORS - SRMS • DEORBIT - SRMS

DIFFERENCES		
ITEM	0020 ORBITER	161-C BASELINE ORBITER
LH_2 TANKS	2 EXTERNAL	1 TWO CELL
LO_2 TANKS	1 TWO CELL-FWD	2 AFT
TUNNEL	SHORT (60 IN.)	LONG (270 IN.)
BODY REF LENGTH	1807 IN.	2178 IN.

Phase 2 of the study.) The separation system for the tank utilizes a pyrotechnically operated piston mechanism, and prior to deorbit, the tanks are spun about a longitudinal axis using solid rocket motors. Deorbit is achieved with a single-rocket motor.

3.3.2 THREE-ENGINE ORBITERS

Three configurations of the three-engine orbiter were selected for conceptual design analyses. These orbiter configurations are identified in Figure 3-8 and include the following:

1. Two-cell main LO_2 tank

2. Two separate main LO_2 tanks

3. A single main LO_2 tank

All of these orbits retain the same structure/TPS and subsystem concepts used in the fully reusable vehicle.

The configuration of the three-engine orbiter with two-cell LO_2 tank is similar to that of the baseline fully reusable system and the two-engine orbiter previously described for the external hydrogen tank concept. This vehicle is illustrated in Figure 3-9.

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0021
(2 CELL LOX TANK)

17 2 FT DIA
71 5 FT
182 5 FT
60 8 FT
156 4 FT
106 2 FT

0018
(2 LOX TANKS)

170 8 FT
63 1 FT
150 0 FT
106.2 FT

0019 "A"
(1 LOX TANK)

170 1 FT
65 0 FT
150 0 FT
106 0 FT
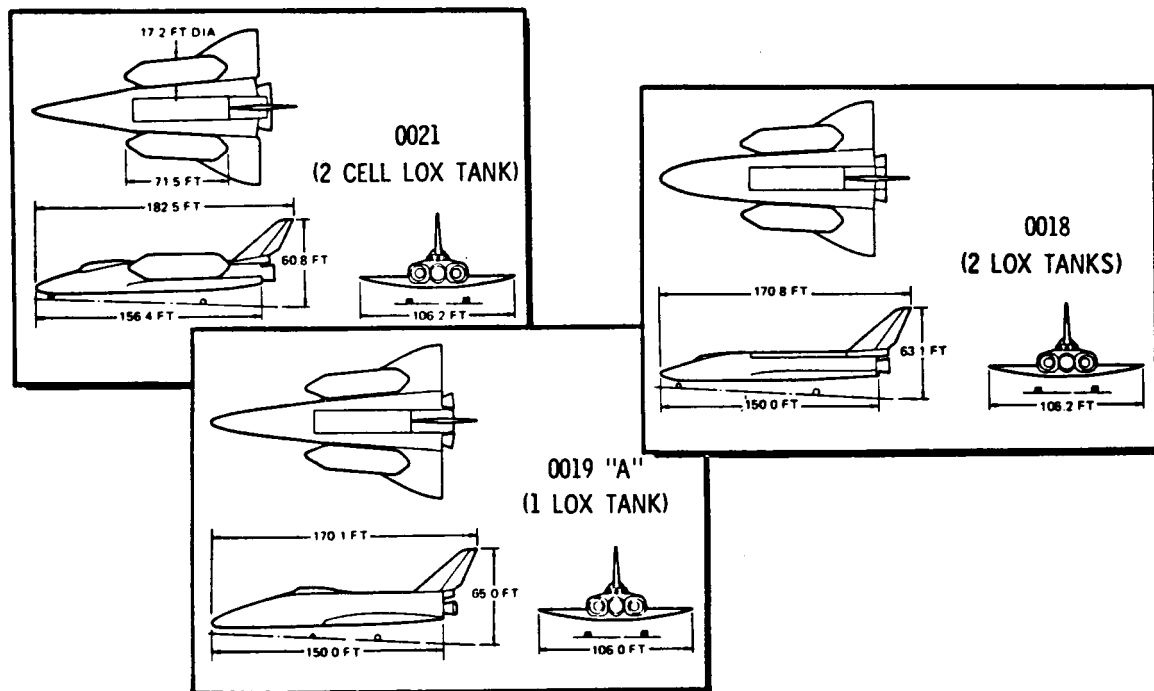
Figure 3-8. Orbiter and External Tankage Assembly,
Three Engine Orbiter, 3012 Cylindrical LH2 Tanks

PAYLOAD BAY
EXTERNAL LH₂ TANKS (2)
OMS ENGINES
EQUIPMENT COMPT
MAIN ENGINES
AIRLOCK/ DOCKING PORT
LO₂ OMS TANKS (2)
LH₂ OMS TANKS (2)
ABES
CREW CABIN
LOX TANK

CHARACTERISTICS	
GROSS WT	● 1,123,449 LB
PAYLOAD	● 40,000 LB
AERO/ THERMAL	● ENTRY L/D = 1.61 ● SUB L/D = 6.4 ● W/C S = 89
PROPULSION SYSTEMS	● MAIN ENGINES = (3) 477 K LB ● ACPS/OMS INTEGRATED TANKAGE ● 29 ACPS THRUSTERS AT 2100 LB EACH ● 3 OMS THRUSTERS AT 10,000 LB EACH ● 4 JTF22A-4 ABES WITH JP FUEL
ORBITER STRUCTURE/ TPS	● LO₂ TANK-2-CELL INTEG ALUM ● PRIMARY – TITANIUM ● HEAT SHIELD – REI
EXTERNAL LH₂ TANK	● ALUM MONOCOQUE ● INTERNAL FOAM INSULATION ● ABLATOR ● SEP SYS PISTONS ● SPIN MOTORS - SRMS ● DEORBIT – SRMS

DIFFERENCES		
ITEM	0021 ORBITER	161-C BASELINE ORBITER
LH₂ TANKS	2 EXTERNAL	1 TWO CELL
LO₂ TANKS	1 TWO CELL-FWD	2 AFT
TUNNEL	SHORT (60 IN.)	LONG (270 IN.)
BODY REF LENGTH	1877 IN.	2178 IN.

Figure 3-9. General Arrangement,
Three-Engine 0021 Orbiter

The aerodynamic characteristics of this system are comparable to the baseline, and the vehicle provides a hypersonic reentry L/D of 1.6 with an angle of attack of 30 degrees. The subsonic L/D is reduced to 6.4 due to the increased base area associated with installation of three main rocket engines.

A configuration directed towards providing greater simplicity is illustrated in Figure 3-10. This configuration provides two separate, circular-section, tapered main LO_2 tanks in the nose of the orbiter. The vehicle is approximately 380 inches less in length than the baseline fully reusable vehicle, and when configured for the polar mission, is approximately 1.1 million pounds at lift-off.

The hypersonic aerodynamic characteristics of this orbiter arrangement are comparable to the reusable system but the wing is moved aft to provide trim capability. The wider forebody of the vehicle results in a large pitching moment at subsonic speeds and therefore requires greater use of aerodynamic surfaces with the resulting reduction in subsonic L/D to 5.7. With these subsonic aerodynamic characteristics, the landing speed is increased to approximately 190 knots.

With a goal of further simplifying the orbiter configuration, a vehicle using a single circular section main LO_2 tank was defined. This vehicle is illustrated in Figure 3-11. Hypersonic aerodynamic characteristics of this vehicle differ from those of the baseline vehicle. The wider nose results in degradation of the maximum hypersonic L/D, and it is therefore necessary for the vehicle to enter at a lower angle of attack (25°) in order to achieve the desired cross range. This vehicle also requires relocation of the wing forward 16 inches relative to the baseline in order to achieve trim capability. Subsonic performance of this vehicle is degraded relative to the fully reusable vehicle due to the increased base drag associated with the three main rocket engines.

3.3.3 CONFIGURATION SELECTION

The previously described orbiter configurations were evaluated to select concepts for comparison with the fully reusable space shuttle system. A summary of this evaluation is presented in Table 3-3. As indicated in the table, the two-engine orbiter using external hydrogen tanks provides aerodynamic performance comparable to the fully reusable system. The combined weight of orbiter and tanks at liftoff is 915,000 pounds compared to the baseline reusable system and has a dry weight of 195,000 pounds. The reduced size of this system results in a reduction in development costs of approximately $200 million compared to the baseline reusable system.

CHARACTERISTICS	
GROSS WT	● 1,123,449 LB
PAYLOAD	● 40,000 LB
AERO/ THERMAL	● ENTRY L/D = 1.61 ● SUB L/D = 5.7 ● W/C_L S = 87
PROPULSION SYSTEMS	● MAIN ENGINES = (3) 477K LB ● ACPS/OMS INTEGRATED TANKAGE ● 29 ACPS THRUSTERS AT 2100 LB EACH ● 3 OMS THRUSTERS AT 10,000 LB EACH ● 2 JTF22A 4 ABES ● 2 JTF22A-4 + AFTERBURNER ABES } WITH JP FUEL
ORBITER STRUCTURE/ TPS	● LO$_2$ TANK-2-SEP INTEG ALUM ● PRIMARY – TITANIUM ● HEAT SHIELD - REI
EXTERNAL LH$_2$ TANK	● ALUM MONOCOQUE ● INTERNAL FOAM INSULATION ● ABLATOR ● SEP SYS PISTONS ● SPIN MOTORS - SRMS ● DEORBIT - SRMS

DIFFERENCES		
ITEM	0018 ORBITER	161-C BASELINE ORBITER
LH$_2$ TANKS	2 EXTERNAL	1 TWO CELL
LO$_2$ TANKS	2 - FWD	2 AFT
TUNNEL	SHORT (60 IN.)	LONG (270 IN.)
BODY REF LENGTH	1800 IN.	2178 IN.

Figure 3-10. General Arrangement, Three-Engine 0018 Orbiter

CHARACTERISTICS	
GROSS WT	● 1,123,449 LB
PAYLOAD	● 40,000 LB
AERO/ THERMAL	● ENTRY L/D = 1.7 ● SUB L/D = 6.3 ● W/C_L S = 125
PROPULSION SYSTEMS	● MAIN ENGINES = (3) 477K LB ● ACPS/OMS INTEGRATED TANKAGE ● 29 ACPS THRUSTERS AT 2100 LB EACH ● 3 OMS THRUSTERS AT 10,000 LB EACH ● 2JTF22A-4 ABES ● 2JTF22A-4 + AFTERBURNER ABES } WITH JP FUEL
ORBITER STRUCTURE/ TPS	● LO$_2$ TANK - INTEG ALUMINUM ● PRIMARY - TITANIUM ● HEAT SHIELD - REI
EXTERNAL LH$_2$ TANK	● ALUM MONOCOQUE ● INTERNAL FOAM INSULATION ● ABLATOR ● SEP SYSTEM PISTONS ● SPIN MOTORS - SRMS ● DEORBIT - SRMS

DIFFERENCES		
ITEM	0019A ORBITER	161-C BASELINE ORBITER
LH$_2$ TANKS	2 EXTERNAL	1 TWO CELL
LO$_2$ TANKS	1 - FWD	2 AFT
TUNNEL	SHORT (30 IN.)	LONG (270 IN.)
BODY REF LENGTH	1800 IN.	2178 IN.
CREW CABIN } EQUIP COMPT }	SEPARATED	SERIES
DOCKING PORT	IN CREW CABIN	IN AIRLOCK

Figure 3-11. General Arrangement, Three-Engine 0019A Orbiter

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SD 71-141-1

Table 3-3. Orbiter Evaluation

FACTOR	2 ENG ORB 2-CELL LO₂ TANK	3 ENGINE ORBITER 2-CELL LO₂ TANK	2 SEP LO₂ TANK	SINGLE LO₂ TANK
OLOW (1000 LB)	915	1,089	1,089	1,089
DRY WT (1000 LB)	195	202	202	203
APPLICATION OF PHASE B BASELINE DATA/CONFIDENCE	HIGH	HIGH	MODERATE	LOWEST
AERO				
HYPERSONIC 40 L/D MAX.	2.3	2.3	2.16	1.88
SUBSONIC 40 L/D MAX.	7.9	6.4	5.7	6.3
ABES	4 JTF22A	4 JTF22A	4 JTF22A + AB	4 JTF22A + AB
TRIM	YES	YES	IMPRACTICAL	YES
TPS ΔWT		0	MINIMAL ≈1,000 LB	MODERATE ≈3,000 LB
MANUFACTURING COMPLEX	MODERATE	MODERATE	LOWEST	LOW-MOD
COST ΔDDT&E $M (RELATIVE TO BASELINE)	-206	-299	-311	-305

SELECTED FOR INTEGRATED VEHICLE COMPARISON

- 2 ENGINE SELECTED FOR COMPARISON USING SAME PROPULSION SYSTEM AS BASELINE ALSO TPS BOOSTER
- 3 ENGINE ORBITER WITH 2 CELL TANK SELECTED
  (1) TO MAINTAIN CONFIDENCE IN CHARACTERISTICS DATA
  (2) COMPARABLE WEIGHTS TO OTHER OPTIONS

Confidence in the predicted characteristics of this vehicle are high, and therefore, the system was selected for comparison with the fully reusable orbiter. The predicted weights for the three-engine orbiters are comparable. The dispersion predicted in development cost reduction for these 3 systems is also small and ranges from $299 to $311 million. Confidence in the characteristics in performance of the three-engine orbiter with the two-cell main LO_2 tank is high, and therefore, this system was selected to provide a good comparison with the fully reusable system.

3.4 CANDIDATE TANK CONCEPTS

The design, weight, and cost of the orbiter external hydrogen tanks are major factors in assessing the merit of this concept. The tank cost would be impacted by system requirements for safe disposal, and therefore, the tank dispersion and probability of impact on shipping or land masses were analyzed. As indicated in Figure 3-12, the allowable dispersion of tanks to prevent impact on land masses is approximately 1400-nm radius. This dispersion would also result in an extremely low probability of an intact tank impacting shipping.

To limit the tank dispersion, it was therefore decided that rocket motors would be used to spin the tank about the longitudinal axis to minimize attitude errors after separation from the orbiter. A number of tank concepts were analyzed, and the system used for vehicle performance and cost analysis in the first phase of the program is illustrated in Figure 3-13. This tank has ablators to maintain an intact system on entry to an altitude less than 200,000 feet. The predicted dispersion with this tank would be a maximum of ± 420 nautical miles.

Upon completion of first phase studies, it was decided that the dispersion of a fragmented task would be acceptable, and therefore, the design for the second phase was configured to allow breakup on entry.

AVG DENSITY
SHIPS/100,000
SQ N MI

HIGH SHIP
DENSITY

$P \approx 3 \times 10^{-7}$
(2 TANKS)

≈ 300 X 300 N MI SQ
≈ 900 X 900
≈ 1500 X 1500
≈ 2100 X 2100
≈ 2700 X 2700

WTR
ETR
1400 N MI R

1980 PROJECTED OVERALL WORLDWIDE MERCHANT
SHIP DISTRIBUTION (INCLUDING FISHING VESSELS)
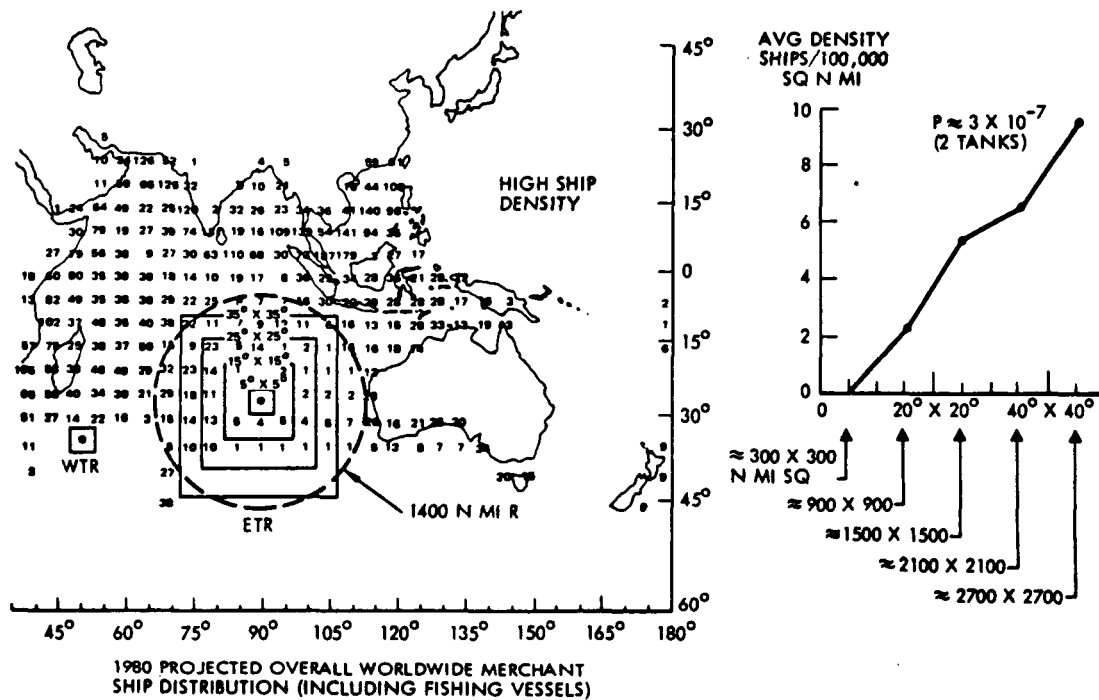
Figure 3-12. Acceptable Impact Area

FORWARD SUPPORT
TRUSS
(2 MEMBERS + PIVOT)

AFT SUPPORT
TRUSS
(TRIPOD–FIXED)

DISCONNECT
PANEL

17 FT

DEORBIT SUBSYSTEM
• SOLID MOTOR
• (126,000 LB/SEC)

INSUL 0.75

CORK

ELECTRICAL DISTRIBUTION
SUBSYSTEM
• POWER
• CONTROL

ROLL SUBSYSTEM
• 4 SOLID MOTORS
• (1320 LB/SEC-WT 8 LB EACH)

THERMAL
PROTECTION
SUBSYSTEM
• SOFI
• CORK

BOOST
REENTRY

PROPULSION SUBSYSTEM
• FEED - 15 IN. D
• PRESSURE 2 IN. D
• VENT
• FILL/DRAIN
• PROPELLANT
  MANAGEMENT
• RECIRCULATION 2 IN. D

STRUCTURE SUBSYSTEM
• TANK
• NOSE FAIRING
• ORBITER ATTACHMENT
• SEPARATION

Figure 3-13. External LH$_2$ Definition

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3.5 CANDIDATE BOOSTER CONCEPTS

This section describes the two booster concepts which correspond with the two- and three-engine orbiters described above.

3.5.1 RERADIATIVE HEAT SHIELD BOOSTER

The booster for the two-engine orbiter system is shown in Figure 3-14. Due to the relatively high staging velocities ($V \geq 9500$ fps) associated with the two-engine orbiters, a reradiative heat shield concept is optimum. The vehicle is basically a scaled-down version of the fully reusable baseline, being approximately 10 feet shorter. It is a low-delta-wing vehicle with a single vertical tail and a small canard surface mounted forward above the body centerline. The body is basically a cylinder with fairings added to streamline the intersection with the aerodynamic surfaces. Since the configurations are similar, the aerodynamic characteristics of this booster are essentially the same as the fully reusable system booster. The wing area is sized to give a landing wing loading at 75.6 psf.

Internally, the vehicle is arranged with the crew compartment and avionics bay located in the nose, a forward LO_2 tank, and an aft LH_2 tank. The tanks are integral tanks and provide the primary load-carrying structure of the booster. The tanks are joined by a cylindrical intertank structure that supports the canards and the forward orbiter attach drag link. All the structural frames are external to the main tanks. The LH_2 tank is internally insulated. A corrugated outer heat shield provides the aerodynamic surface of the body.

The main propulsion system consists of eleven 550,000 pound sea-level thrust LO_2/LH_2 engines installed in the base or the vehicle. Ten turbofan engines are deployed below the body and wing for flyback propulsion. The other vehicle subsystems are similar to the fully reusable baseline with only minor changes due to vehicle size.

3.5.2 HEAT SINK BOOSTER

As previously discussed, the use of a three-engine orbiter results in staging velocities less than 8000 fps which makes the heat sink booster an attractive concept. The heat sink booster developed in the first phase of this study is shown in Figure 3-15. This booster was derived from the fully reusable system booster and has a low delta wing, a single vertical tail and

STAGING VELOCITY   9,922 FPS
BOOSTER GLOW   3,827,000 LB
LANDING W/S   75.6 LB/FT

60°

53°

134.6 FT.

1.5 FT.

TANK DIA.
31.62 FT.

LO_2 LH_2

LANDING C.G.

98 FT.

8.0
CLEAR

(11)
550 K SL MAIN
ENGINES
(ϵ-55)

STA.
1000   STATIC GROUND LINE 2° 40 FT.   STA.
3808

15° GIMBAL-PLANE

234.0 FT. (REF. LENGTH)
258.0 FT.

38 FT.

(10)
JTF 22A-4
FLYBACK-
ENGINES

Figure 3-14.   B-9W Booster Reradiative Heat Shield System
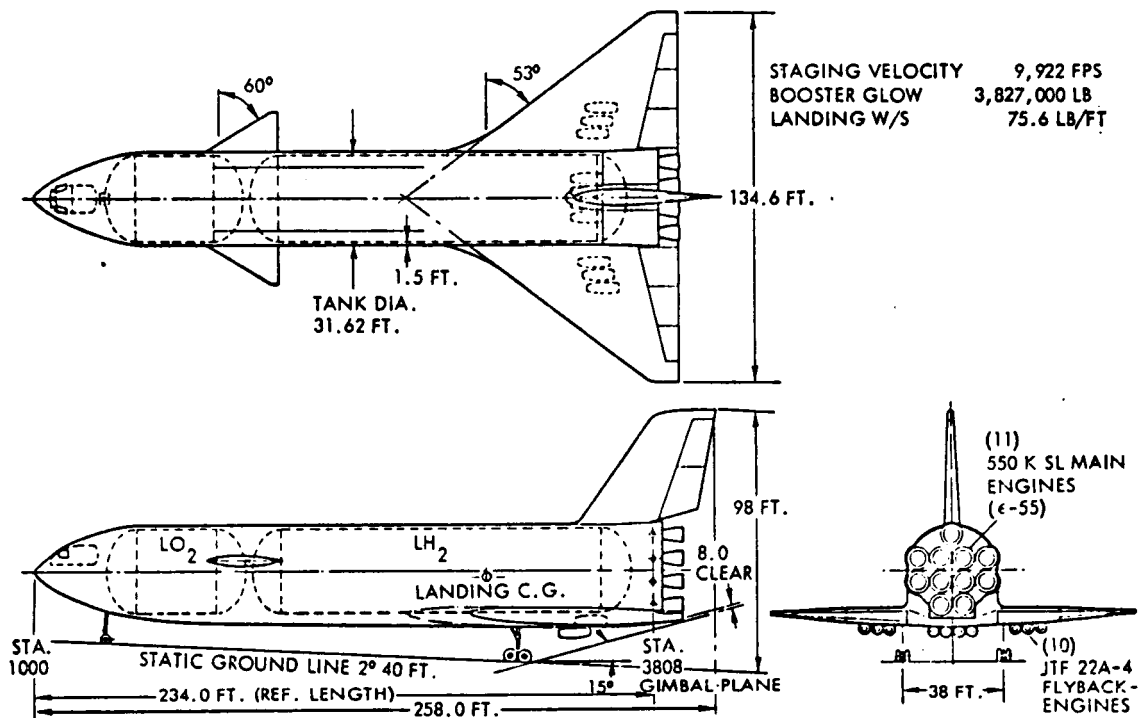
STAGING VELOCITY   7,719 FPS
BOOSTER GLOW   2,881,000 LB.
LANDING W/S   75.6 LB.FT.^{2}

53°

60°

LO_2 LH_2

126.5 FT.

11%

O/D 30.55 FT.
I/D 30.50 FT.

O/D 30.55 FT.
I/D 30.00 FT.

(13)
415K SL MAIN
ENGINES
(ϵ-35)

35°

75% REF.
LENGTH

93 FT.

HOLD DOWN

30°

65°

15°

209.92 REF. LENGTH

STA. 1000   STA. 3519 GIMBAL PLANE

(8)
JTF22A-4
FLYBACK
ENGINES

Figure 3-15.   B-17D Booster Heat Sink System

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a forward-mounted canard. Due to the lower propellant load requirement, this booster is about 40 feet shorter than the fully reusable baseline. Being a heat sink vehicle, most of the reradiative heat shield cover panels on the body are eliminated, and the walls of the LO_2 tank, intertank, and LH_2 tank provide the aerodynamic surface. A fairing is necessary in the area of the wing body intersection which includes a ramp to go from the lower surface of the LH_2 tank to the lower wing surface. The canard fairing is retained to prevent gap heating. The LO_2 lines and other subsystem lines must be routed external to the tanks on the upper vehicle surface.

The subsonic and hypersonic aerodynamic characteristics of this vehicle were estimated to be somewhat better than the reradiative heat shield booster due to a greater ratio of exposed wing area to body area and the elimination of most of the corrugated body surface.

Internally the vehicle arrangement is similar to the reradiative heat shield vehicle. The aluminum wall thickness of the LO_2 tank, intertank, and LH_2 tank is increased above the thickness required for strength as necessary to provide sufficient heat sink material. The stiffening material and structural frames for the tanks and intertank are located internally to yield a smooth surface.

The delta wing is a titanium heat sink on the lower surface with a hot titanium upper surface. The canard and vertical are both heat sink construction.

The main propulsion system consists of thirteen 415,000-lb sea-level thrust LO_2/LH_2 rocket engines installed in the base of the vehicle. Eight turbofan engines are deployed below the wing and body for flyback propulsion. The elimination of a large portion of the reradiative heat shield on the body reduces significantly the requirement for purging. The other vehicle subsystems are similar to the fully reusable baseline with only minor changes due to smaller vehicle size and shorter flight time.

3.6 REUSABLE SHUTTLE

To facilitate a comparison with the orbiter external hydrogen tank concept, a number of reusable space shuttle systems were synthesized. The sizes of the reusable shuttle using a two-engine orbiter were defined with the vehicle sized to account for the impact of ground winds on booster flyback propellant. The system also utilizes orbit maneuvering engines operating in parallel with the main propulsion system. This system and a reusable vehicle using a three-engine orbiter are defined in Table 3-4. For a three-engine orbiter, the logistic mission becomes critical in establishing the vehicle size. As shown in Table 3-4, a gross lift-off for the system with two-engine orbiters is approximately 4.8 million pounds and 4.3 million pounds for the three-engine orbiter system.

Table 3-4. Fully Reusable Size Revisions

		Configuration		
		ϕ B Baseline	$N_B/N_o = 12/2$ $F_{SL} = 550K$ Lb	$N_B/N_o = 14/3$ $F_{SL} = 415K$ Lb
Critical sizing mission		Polar	Polar	Logistic
GLOW	(1000 lb)	5047	4872	4374
T/W		1.308	1.355	1.328
V_S (rel)	(fps)	10832	10486	7770
BLOW	(1000 lb)	4188	3980	3148
Dry weight	(1000 lb)	627	608	489
Landing weight	(1000 lb)	639	620	497
OLOW	(1000 lb)	859	892	1225
Dry weight	(1000 lb)	224	227	275
Landing weight	(1000 lb)	268	272	307
Payload	(1000 lb)			
Due east		81.7	≈80	≈70
55 deg		36.8	≈36	25
Polar		40	40	≈43

Bullet list inside table:
- Updated orbiter and booster weights and booster flyback range
- OMS ascent assit and directional booster flyback SINDS
- Computer-synthesized vehicles

3.7 CONFIGURATION EVALUATION AND SELECTION

Phase 1 studies indicated that with the orbiter external hydrogen tank concept, the three-engine orbiter configuration resulted in the lowest weight and lowest cost system. This three-engine orbiter also facilitated use of a heat sink booster due to the lower staging velocity. Comparable weights and costs were estimated for the candidate orbiter configurations analyzed. The configuration with high fineness ratio body similar to that of the fully reusable vehicle provided the greatest confidence, and therefore, it was selected for comparison with the fully reusable system.

The Phase 1 studies also showed that significant weight and cost reduction could be achieved in the fully reusable space shuttle through incorporation of three engines on the orbiter. This three-engine orbiter also allowed use of a heat sink booster in the fully reusable space shuttle concept.

A comparison between the OEHT system and the fully reusable vehicle using two- and three-engine orbiters is presented in Table 3-5. As indicated in the table, the external hydrogen tank concept, using a three-engine orbiter, results in approximately 400,000 pounds lower gross lift-off weight than the fully reusable system. The development cost for this system is also lower than for the fully reusable vehicle.

Based on Phase 1 studies, however, the fully reusable vehicle has a projected lower program cost than the orbiter external hydrogen tank concept. As illustrated by the cumulative costs defined in Figure 3-16, the orbiter external hydrogen tank concept results in higher program costs than a fully reusable system after approximately 300,000 flights.

A number of additional factors to be considered in preparing the orbiter external hydrogen tank concepts with the fully reusable vehicle are: (1) impact of fracture mechanics on LH_2 tank design, (2) revised system purge requirements, and (3) abort capability of the two concepts analyzed.

To facilitate a good comparison between the OEHT concept and the fully reusable system, it was decided that both concepts should be analyzed during the second phase of the study. The selected OEHT concept consists of a three-engine orbiter with two-cell main LO_2 tank and cylindrical external LH_2 tanks plus a heat sink booster. The selected reusable vehicle for further study uses a three-engine orbiter and heat sink booster. Analysis on the reusable system during Phase 2 is limited to definition of the vehicle size and configuration.

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Table 3-5. OEHTS and Reusable Vehicle Comparison

| Factor		External LH$_2$ Tank		Reusable System	
		2-Engine Orbiter	3-Engine Orbiter	2-Engine Orbiter	3-Engine Orbiter
Integrated Vehicle					
GLOW Polar	(1000 lb)	4,716	4,004	4,872	4,368
PL Polar	(1000 lb)	40	40	40	≈43
PL 55° inclin	(1000 lb)	≈34	≈25	≈36	25
PL 28.5° inclin	(1000 lb)	≈73	≈67	≈80	≈70
Vs Polar	(fps)	9,922	7,719	10,486	≈7,800
Orbiter					
Concept		2-cell LO$_2$	2-cell LO$_2$	2-cell LH$_2$	2-cell LH$_2$
OLOW Includes tanks	(1000 lb)	889	1,123	892	1,225
Dry weight Excludes tanks	(1000 lb)	197	215	227	275
Tanks					
Concept		Cylind w/ ablator	Cylind w/ ablator	None	None
Lift-off weight	(1000 lb)	105	140		
Dry weight	(1000 lb)	21	25		
Booster					
Concept		TPS	Heat sink	TPS	Heat sink
BLOW	(1000 lb)	3,827	2,881	3,980	3,148
Dry weight	(1000 lb)	578	454	608	489
Operation			Reduced Maint		Reduced Maint
DDT&E cost	($M)	6,543	6,298	6,555	6,367
First unit cost	($M)	400	393	407	410
Tank cost	($M)	355	453	-	-
Program cost, 445 flights	($M)	9,488	9,268	9,242	9,056
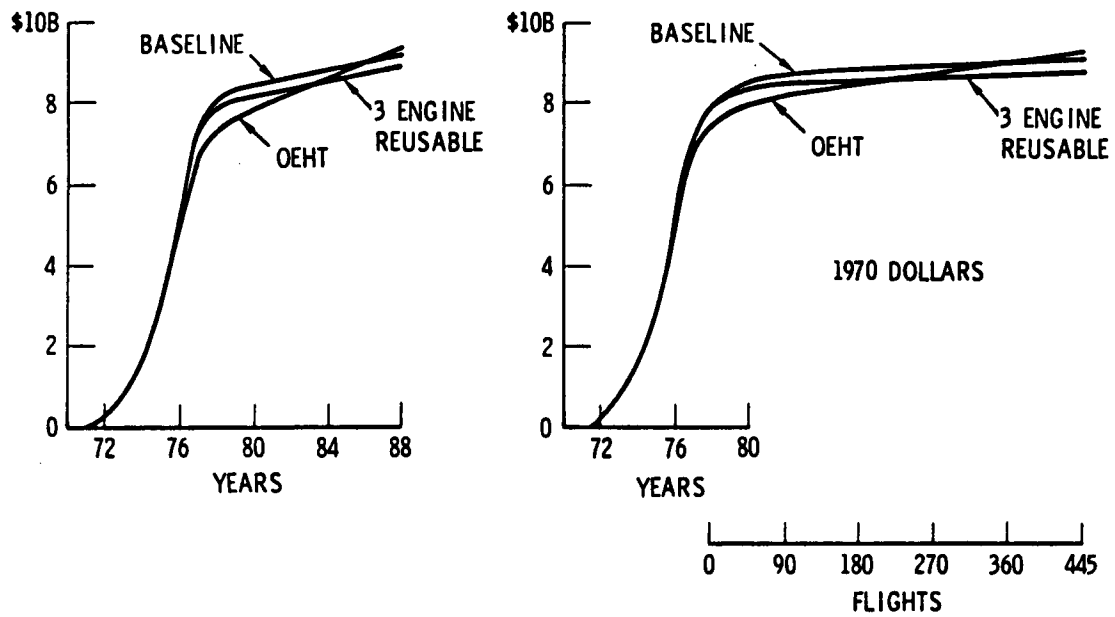
Figure 3-16. Program Cost Comparison

PHASE 2

4.0 SELECTED VEHICLE DEFINITION

Preliminary design analyses during the second phase of the study were performed on a single orbiter external hydrogen tank space shuttle configuration identified as the lowest cost system during Phase 1 of the program. This system uses a three-engine orbiter and heat sink booster with 13 main engines, each delivering 415,000 pounds sea level thrust. Based on the preliminary design analysis of the selected configuration, the subsystem definitions and weights were updated. This section of the report describes the configuration and subsystems of the selected vehicle and the associated analyses and trade studies. Initial weights computed in a preliminary vehicle synthesis are provided as are the updated weights based on detailed subsystem analyses. The impact of these revised subsystem weights on the vehicle size required to accomplish the design reference missions is reported in Section 4.5. For comparison purposes, the two and three - engine, fully reusable space shuttle vehicles are presented in Section 4.6.

4.1 INTEGRATED VEHICLE

The configuration of the orbiter external hydrogen tank space shuttle, selected as a result of Phase 1 studies, did not change significantly due to the preliminary design analysis. Trade studies were performed to establish the desired position of the orbiter on the upper surface of the booster and also to assess a number of tank to orbiter aerodynamic fairing concepts for the mission boost phase. These trade studies influenced the orbiter aerodynamic fairing design but had minimal effect on the outward appearance of the shuttle for the boost phase of the mission.

The integrated vehicle subjected to the preliminary design analysis is illustrated in Figure 4-1. This vehicle, with a gross liftoff weight of 4,004,000 pounds, does not reflect a size revision which would be required to reflect the subsystem weights analysis.

4.1.1 VEHICLE DESIGN AND MASS PROPERTIES SUMMARY

As illustrated in Figure 4-1, the vehicle consists of a delta wing booster with 13 main engines each delivering 415,000 pounds sea level thrust and a delta wing orbiter using three engines with the same powerhead as that used on the main propulsion system for the booster. Due to the low staging velocity associated with this system, the lowest weight booster is achieved using a structure heat sink for thermal control. The booster vehicle is similar in shape to that used on the fully reusable system defined as a

baseline in Phase B studies. It has a low delta wing, single vertical tail, and forward mounted canard.

The orbiter is similar in aerodynamic shape to the baseline all-reusable shuttle orbiter and has a delta wing and high-fineness-ratio body. The internal tankage arrangement is, however, revised in comparison to the fully reusable orbiter, and structural modifications reflect the requirements for attachment of external hydrogen tanks.

The location of the orbiter on the booster is similar to that for the fully reusable space shuttle and provides comparable clearance between the aft end of the orbiter and the booster center vertical. Due to the reduced vehicle weight and resulting decreased booster length, the orbiter overhang relative to the booster is increased over that of the fully reusable system. Attachment of the orbiter to the booster is achieved through a linkage system identical in design to that used for the fully reusable shuttle. This will result in separation dynamics comparable with those of the fully reusable shuttle. Attachments of the links on the booster are revised from those for the fully reusable system, and additional frame supports are required within the booster LO_2 tank to accommodate the forward link attachment. In order to identify the most desirable location of orbiter on booster, a trade study was performed to assess the relative merit of supporting the orbiter in the mid-body section compared to providing orbiter attachments at the nose of

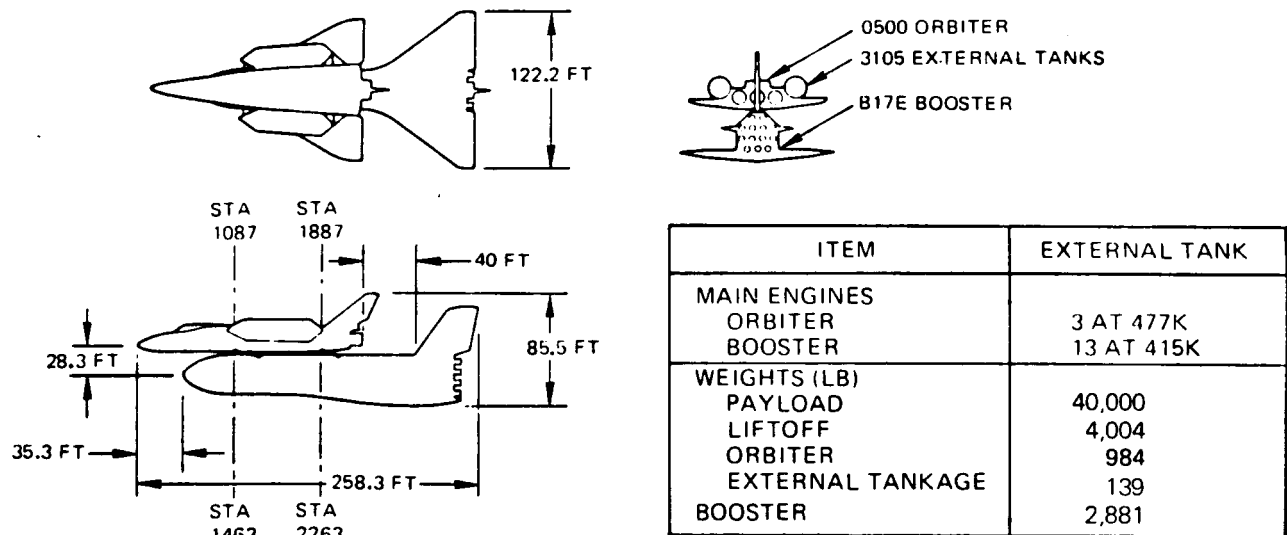| ITEM	EXTERNAL TANK
MAIN ENGINES	
ORBITER	3 AT 477K
BOOSTER	13 AT 415K
WEIGHTS (LB)	
PAYLOAD	40,000
LIFTOFF	4,004
ORBITER	984
EXTERNAL TANKAGE	139
BOOSTER	2,881

Figure 4-1. Integrated Delta Vehicle

the vehicle. This trade study indicated that vehicle weights would be increased with an orbiter nose support concept and, therefore, the baseline system previously described was selected.

The preliminary mass properties of the selected baseline orbiter external hydrogen tank integrated vehicle are defined in Table 4-1. These mass properties are based upon the initial vehicle synthesis performed during the first phase of the program. Subsequent subsystem analyses resulted in weights revisions for both the orbiter and booster. The revisions and their impact on the weight contingency are also identified in the table. The impact of these updated weights on the vehicle size required to perform the mission with a 10- percent weight contingency in orbiter and booster are identified in Section 4.5.

4.1.2 AERODYNAMICS AND STABILITY

The major aerodynamic considerations for the orbiter external hydrogen tank space shuttle during the ascent phase of the mission are the impact of the external tanks on vehicle aerodynamic stability and boost drag. As illustrated in Figure 4-2, the center of gravity is aft the integrated vehicle aerodynamic center in both pitch and yaw over the ascent mach number range. This system therefore has aerodynamic stability during booster operation and has the advantage that loss of thrust vector control will not result in diverging vehicle dynamic motion. It will also result in an inherent reduction in angle of attack as the vehicle is flown through a wind profile or gust. The mated vehicle drag coefficient (C_A $\alpha = 0$) is also presented in Figure 4-2 and indicates that the ascent drag coefficient of the vehicle configuration is increased by approximately 40 percent due to the effects of the external tanks. The drag data presented in Figure 4-2 for the contribution of external tanks is extracted from aerodynamic wind tunnel tests performed on the Grumman/Boeing wind tunnel model and add to the data compiled during recent NR/GD wind tunnel tests at Ames Research Center on the reusable baseline vehicle.

4.1.3 ASCENT TRAJECTORIES AND PERFORMANCE

The selected vehicle concept using a 13-engine booster and three-engine orbiter (described in Section 4.1) was analyzed to establish the trajectory which would maximize vehicle performance and minimize gross liftoff weight. Based on weights associated with the detailed subsystem analysis, the specified OEHT concept was resized to satisfy the requirements of the three design reference missions. The critical vehicle sizing mission is delivery of a 40,000-pound payload into a 100-n.mi. polar orbit. Sizing of the vehicle to satisfy this mission requirement results in a vehicle liftoff weight of 3,896,000 pounds. Characteristics of the point mass trajectory

Table 4-1. Sequence Mass Properties Statement

FORM 3945 A-2 NEW 8-70

CONFIGURATION						BY NR		DATE		PAGE OF	
MODEL 0500/B-17E ORBITER/BOOSTER			CENTER OF GRAVITY INCHES			MOMENT OF INERTIA SLUG FT² X 10⁻⁶			PRODUCT OF INERTIA SLUG FT² X 10⁻⁶		
NO.	MISSION EVENT	WEIGHT LB.	X	Y	Z	I_{xx}	I_{yy}	I_{zz}	I_{xy}	I_{xz}	I_{yz}
*	LIFTOFF	3983723	1803	0	468	33.588	372.422	363.282	0	-34.641	0
*	MAX Q FLIGHT CONDITION	3005923	1892	0	491	31.660	334.147	326.348	0	-40.128	0
*	3 G FLIGHT CONDITION	1967431	1927	0	539	28.034	289.426	284.479	0	-42.142	0
*	BOOSTER BURNOUT	1661518	1853	0	553	26.640	252.367	247.700	0	-40.925	0
*	BOOSTER LESS ORBITER	538069	2729	0	273	3.481	62.37	63.79	0	- 0.329	0
*	BOOSTER START CRUISE	522019	2726	0	271	3.360	60.97	62.47	0	- 0.167	0
*	BOOSTER LANDING	463404	2760	0	275	3.232	54.23	55.77	0	0.358	0
**	ORBITER - LIFTOFF	1123448	1105	0	356	9.661	43.608	51.012	0	1.205	0
**	ORBITER - BURNOUT	314002	1507	0	366	3.285	21.889	24.067	0	0.040	0
**	ORBITER - ON ORBIT	292788	1513	0	359	2.793	21.422	23.242	0	0.088	0
**	ORBITER - RE-ENTRY	259806	1515	0	363	2.484	19.180	20.739	0	0.182	0
**	ORBITER - LANDING	258556	1514	0	360	2.432	18.730	20.436	0	0.132	0

NOTES: ● BOOSTER REFERENCE DATUM
BOOSTER NOSE STATION = 1000
BOOSTER TANK CENTERLINE = 400
BOOSTER LENGTH (NOSE TO GIMBAL POINT) = 2400 INCHES

** ORBITER REFERENCE DATUM
ORBITER NOSE STATION = 200
ORBITER PAYLOAD BAY CENTERLINE = 400
ORBITER REFERENCE LENGTH = 1940

UPDATED DATA

PITCH AC
(PERCENT BOOSTER LENGTH)

LONGITUDINAL CHARACTERISTICS

- PITCH STATIC STABILITY OVER MACH RANGE

AC

c.g.

MACH NO.

FOREBODY
$CA_\alpha = 0$

$CA_F \ \alpha = 0$ MATED (TOTAL) CONFIGURATION

$CA_F \ \alpha = 0$ DUE TO ORBITER & BOOSTER

$CA_F \ \alpha = 0$ DUE TO TANKS

MACH NO.

YAW AC
(PERCENT BOOSTER LENGTH)

DIRECTIONAL CHARACTERISTICS

- YAW STATIC STABILITY OVER MACH RANGE

AC

c.g.

MACH NO.

DATA SOURCE

- NR BASELINE B9U/161C ORBITER/BOOSTER
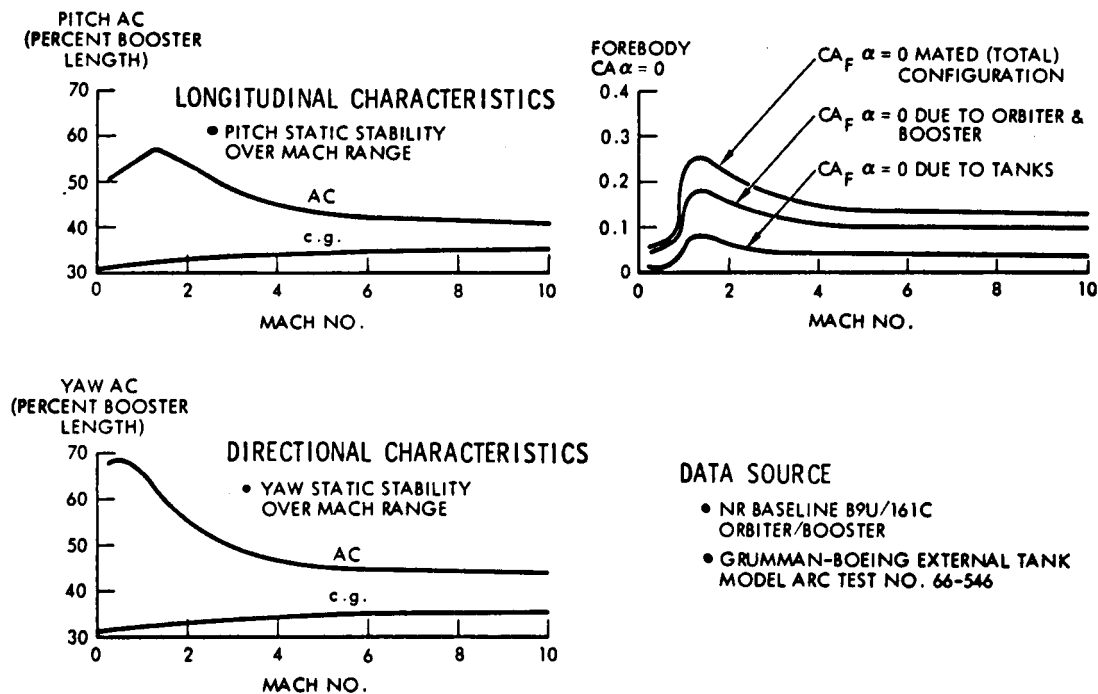- GRUMMAN-BOEING EXTERNAL TANK MODEL ARC TEST NO. 66-546

Figure 4-2. Integrated Vehicle Aerodynamics

with zero lift are presented in Figure 4-3. This trajectory is based on a vehicle liftoff thrust-to-weight ratio of 1.39 and results in a maximum dynamic pressure of 573 psf. In order to limit the ascent acceleration to 3 g, the booster main rocket engines are throttled by 17 percent. Staging of orbiter from booster occurs at an altitude of 201,450 feet with a flight path angle of 16 degrees and a relative velocity of 7,333 fps. This staging condition reflects vehicle optimization considering wind direction affects on booster fuel requirements for flyback. The booster flyback range is approximately 213 miles. Following staging, the orbiter is injected into a 50 by 100 n. mi. orbit. To limit the ascent acceleration to 3 g. (nominal) during orbiter operation, the orbiter main rocket engines are throttled by 36 percent. The impact of aerodynamic lift and the control system requirements on the trajectory were also established. The control system impact on ascent environment is defined in following sections.

As stated, the vehicle was sized to meet the requirements of the polar mission. Therefore, the vehicle performance capability for the three design missions was determined in order to verify that the vehicle also provides the required payload capability for the due-east launch mission and the logistic resupply mission. The performance for the three design missions is shown in Table 4-2.

- FINAL OEHT VEHICLE SIZING
- GLOW = 3896K LB
- N_B/N_O = 13/3
- F_{SL} = 415K LB

- $\bar{q}_{MAX}$ = 573 PSF
- STAGING,
  - V_r = 7333 FPS
  - h = 201,450 FT
  - γ_r = 16 DEG
  - $\dot{\bar{q}}$ = 15 PSF
- F/B RANGE = 213 N MI

17% THROTTLE   36 PERCENT THROTTLE

AXIAL ACCELERATION, (T-D)/W (G'S) — DYNAMIC PRESSURE, $\bar{q}$ (PSF) — REL VELOCITY, V_r (K FPS) — ALTITUDE, h (K FT) — REL FLIGHT PATH ANGLE, γ_r (DEG)

TIME FROM LIFT OFF (SEC)
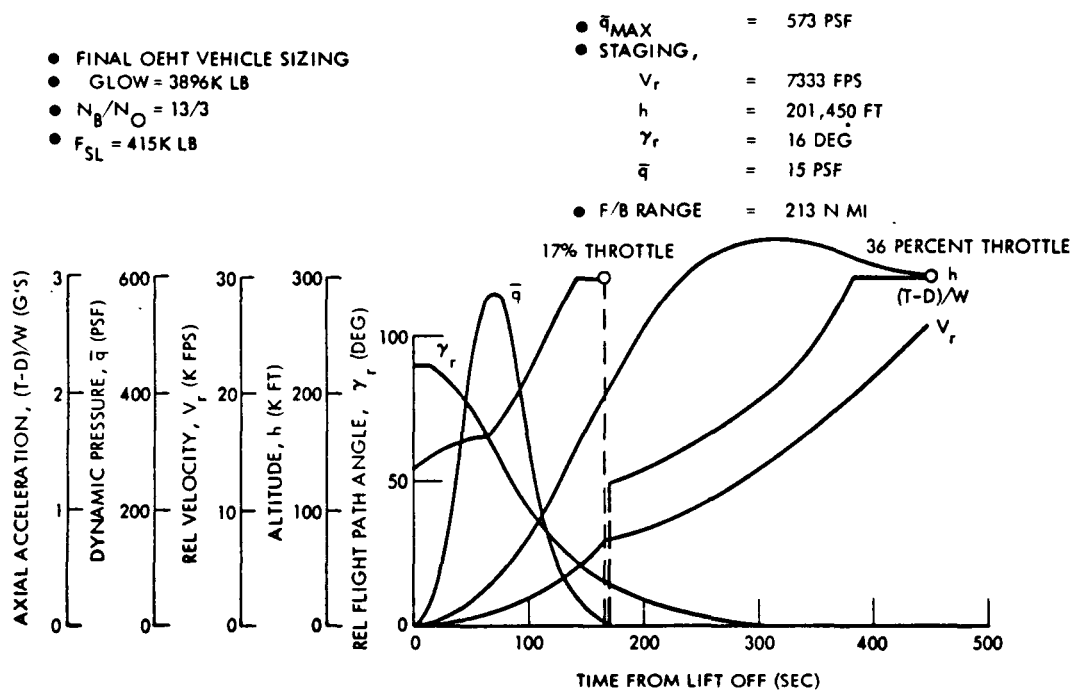
Figure 4-3.   Final OEHT Vehicle Sizing, GLOW = 3,896,000 Pounds

Table 4-2.   Alternate Mission Performance

Mission/ Inclination	Mission On-orbit ΔV (fps)	Orbiter ABES	OMS Ascent Assist	Booster Staging			Payload Capability (1000 lb)
				V_r (fps)	h (1000 ft)	γ_r (deg)	
Due east -28.5°	900	Out	No	7634	202.5	14	67.7
Logistic -55°	1500	In[1]	No	7795	194.6	12	25.4
Polar -90°	650	Out	Yes	7719	206.6	14	40.0

(1) ABES weight = 18,500 pounds

4.1.4 ASCENT CONTROL

Ascent performance and control analyses were conducted to determine the effect of lifting trajectories and the ascent control mode that would minimize the vehicle gross liftoff weight for the design reference missions.

The performance and trajectory optimization simulation program (Honeywell) includes the penalities associated with booster flyback propellants and the OMS propellants required for abort. Trajectory concepts studied were gravity turn and zero angle of attack. The results are shown in Figure 4-4 where the mass injected into orbit is plotted against staging attitude. Results of the NR vehicle synthesis program also are shown. Of the two trajectory types flown by Honeywell, the zero angle-of-attack concept is better by approximately 2600 pounds payload, injecting 318,500 pounds into a 50 by 100-n.mi. polar orbit and staying within the design $q\alpha$ and $q\beta$ limits.

In evaluating the various ascent control concepts, their impact on mission ideal velocity requirement and ascent loads with the associated impact on structure weight was considered. Three control concepts were analyzed. They are:

1. Attitude control or programmed flight path angle

2. Attitude control with minimum drift to limit deviations from a nominal trajectory

3. Attitude control with load relief through the sensing of vehicle acceleration and attitude rates to control the vehicle and limit its angle of attack.

The control analyses investigated the effect of steady state winds and gusts and head, tail, and side wind directions. Wind gusts at altitudes of 12,600 feet were selected as representative cases to establish the maximum main rocket engine gimbal requirements. The goal for all control concepts was to limit structural loads to those of the baseline fully reusable system in which max $q\alpha$ = 2800 psf degrees and max $q\beta$ = 2400 psf degrees. A constraint of ±10 degrees on main rocket engine gimbal was also imposed, and the effect of two main rocket engine failures was considered. Figure 4-5 illustrates the effect of the three previously specified control concepts on the ascent $q\alpha$ and $q\beta$ and bank angle. As shown, the use of minimum drift or load relief results in $q\alpha$ and $q\beta$ values which result in loads below the structural load-carrying capability of the baseline and OEHT systems.

The load relief control concept is preferred over the minimum drift concept, as it was for the fully reusable baseline vehicle, because of the lower $q\alpha$ and $q\beta$.

The required main rocket engine gimbal requirement for the booster ranges from +8 degrees following failure of the two top engines at staging to -10 degrees for booster-only flight.
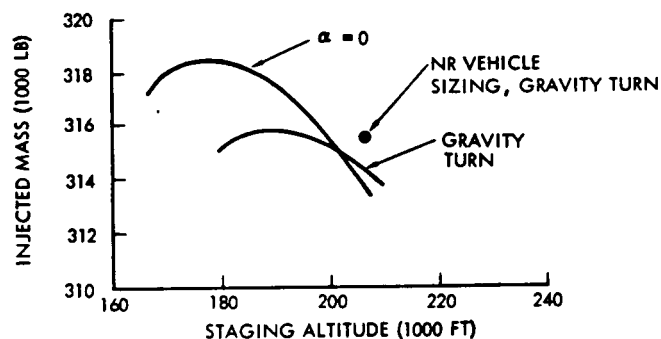
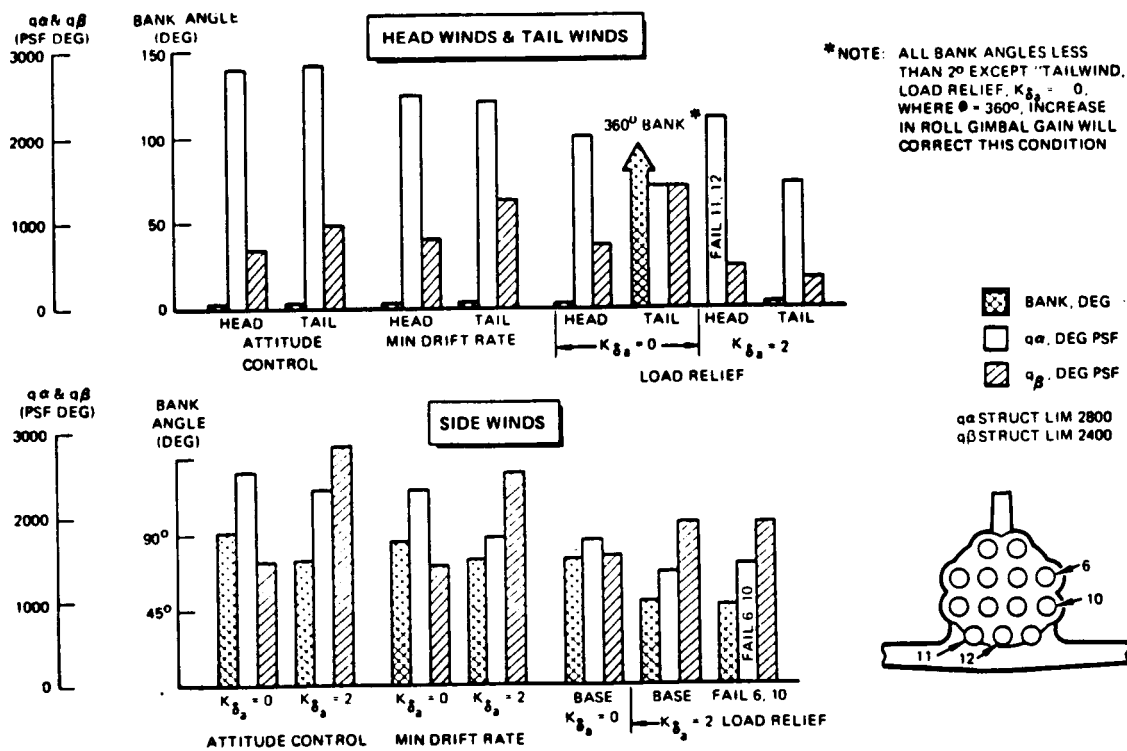Figure 4-4.   Mass Injected Into Orbit

Figure 4-5.   Effect of Control Concept on Loads and Responses

4.1.5 VEHICLE LOADS

Structural loads data were developed for the 0500 orbiter in support of the study of the external tank orbiter configuration. The analysis configuration is based on Dwg. VB70-0500. The orbiter/booster forward and aft attachment stations are 1087 and 1887, respectively. The longitudinal (X) attachment load is carried at the forward attachment. The vertical (Z) loads due to the external LH_2 tanks are reacted at Stations 1087 and 1887. The external tank longitudinal (X) load is reacted at Station 1939. Body load distributions are referenced to WL 400 at the plane of symmetry. The 0500 orbiter weight data for the loads analysis are summarized here.

Liftoff weight	=	983,800 pounds
External tanks	=	148,000 pounds
Total at liftoff	=	1,131,800 pounds
Flyback weight	=	260,000 pounds
Payload	=	40,000 pounds

Using these values, estimated weight distributions were developed for the structural loads analysis. The maximum $q\alpha$ boost condition is based on the 161C baseline criteria: $q\alpha = \pm2800$ psf-degrees.

Body load distributions were developed for the following conditions:

High-Q boost head wind

High-Q boost tail wind

Booster end-burn

Orbiter end-burn

Subsonic maneuver

Two-point landing

Three-point landing

The orbiter/booster attachment fitting loads are summarized in Figure 4-6. These are the total loads at each attachment station and must

EXTERNAL TANK LOAD X 10⁻³ ATTACHMENT (PER SIDE)			
	FWD VERT	AFT VERT	AFT DRAG
HI Q HEADWIND	-226	13.4	-372
HI Q TAILWIND	-143	-13.6	-307
BOOSTER END BURN	19.3	19.3	-244
ORBITER END BURN	1.6	1.6	-54.4

VERTICAL TAIL ROOT		
COND	SHR X 10⁻³	M X 10⁻⁶
HI Q	±114	±15.4
SUBSONIC MANEUV	±25	±3.4

WING ROOT LOADS		
CONDITION	SHR X 10⁻³	M X 10⁻⁶
HI Q HEADWIND	-278	46.5
HI Q TAILWIND	364	-60.7
SUBSONIC MANEUVER	-190	31.7

ORBITER/BOOSTER ATTACH LOADS			
CONDITION	STA 1087		STA 1887
	$F_X \times 10^{-3}$	$F_Z \times 10^{-3}$	$F_Z \times 10^{-3}$
HI Q HEADWIND	3269	659	154
HI Q TAILWIND	3006	782	-1254
BOOSTER END BOOST	3742	-44	-486

NOSE GEAR LANDING LOAD X 10³			
CONDITION	VERTICAL	DRAG	LATERAL
THREE PT SPIN UP	45.6	35.1	0
THREE PT SPRING BACK	50.7	-34.8	0
THREE PT BRAKED ROLL	70.6	0	0
UNSYM BRAKING	41.9	0	16.5

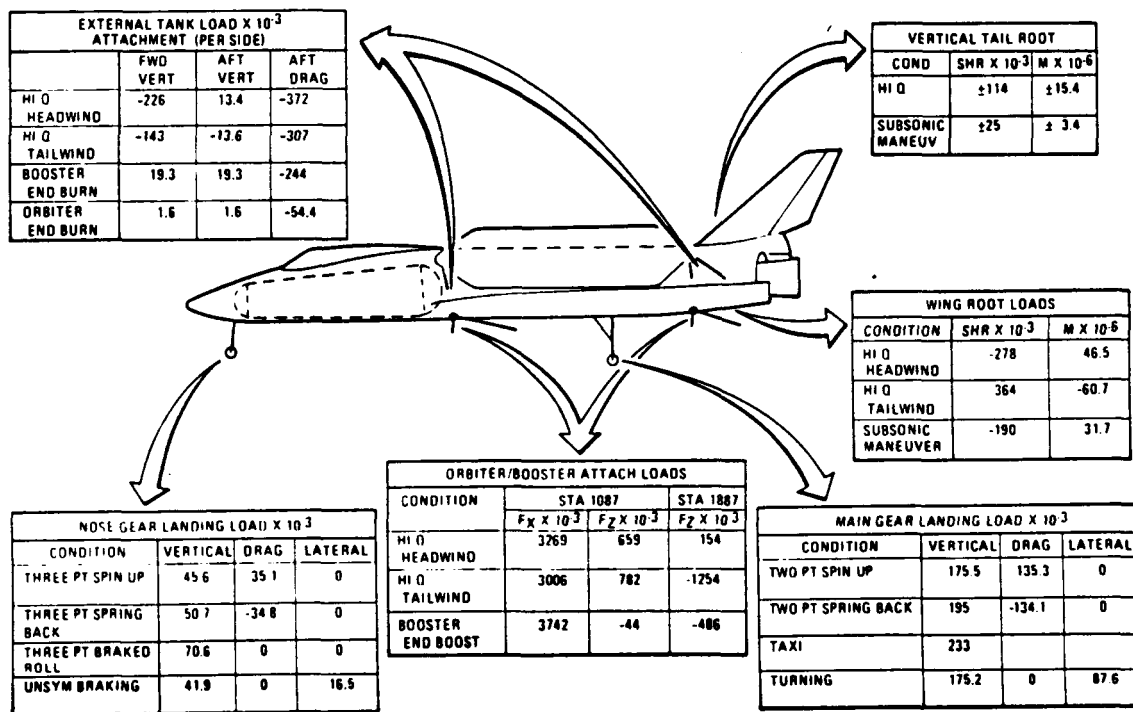| MAIN GEAR LANDING LOAD X 10³			
CONDITION	VERTICAL	DRAG	LATERAL
TWO PT SPIN UP	175.5	135.3	0
TWO PT SPRING BACK	195	-134.1	0
TAXI	233		
TURNING	175.2	0	87.6

Figure 4-6. Loads

be distributed to the individual fittings. The critical external tank attachment fitting loads are included in the figure material. These loads are shown for one side.

Loads data for the wings, vertical tail, and landing gear are also given. Aerodynamic pressure distributions for the external tank nose fairings were determined. These are shown in Section 5.2.5, Volume II, as are the orbiter body bending moment, shear, and axial loads.

4.1.6 ASCENT THERMAL ENVIRONMENT

The ascent thermal environment was established based on the ascent flight trajectory and mated vehicle configuration characteristics. The booster, orbiter, and tank design thermal environment were established considering the total mission, ascent, and reentry. These data support the booster, orbiter, and tank design studies and weight analysis. The design thermal environment and TPS requirements are presented in Sections 4.2.6, 4.3.6, and 4.4.6 for the orbiter, tanks, and booster, respectively.

4.1.7 ABORT CAPABILITY

Requirements for safe mission termination with the OEHT concept are the same as those for the fully reusable baseline and require intact abort capability. In the event of mission termination prior to liftoff, crew members egress to an elevator on the launch tower and subsequent transportation to a safe room.

Requirements for safe mission termination also are imposed following failure of a main rocket engine on either booster or orbiter. Following a failure of a main rocket engine on the booster, the OEHT Concept may accomplish the primary mission with overthrust on remaining operating rocket engines in the same manner as the fully reusable vehicle. Following a main rocket engine failure on the orbiter, the abort mode is to continue in a once-around abort trajectory.

While requirements for early separation prior to booster propellant depletion have not been identified, the abort capability of the OEHT concept for early separation has been analyzed. The analysis has considered the need for safe recovery of booster and orbiter to launch site following early staging. This requires that a satisfactory trajectory be identified in which both orbiter and booster can return to the launch site without exceeding their structural load-carrying capability, thermal protection system environmental constraints, and the capability of their control systems. The safe mission termination also requires that the vehicle be capable of operating in high dynamic pressure regimes. To establish the abort capability of the OEHT concept, the recovery capability was established for separation at 20 seconds, 80 seconds, 169 seconds and 176 seconds after liftoff. The results of this abort analysis are summarized in Table 4-3 in which the capabilities of the OEHT concept are compared to those of the baseline fully reusable shuttle.

As indicated in the table, it has been established that both orbiter and booster can return to the launch site following early separation without exceeding the structural load-carrying capability or thermal constraints associated with their designs. Preliminary analysis also indicates that the vehicles will be controllable. The trajectory constraint imposed by the thermal protection limitations of the external hydrogen tank have not been analyzed. To achieve the specified abort capability, it may be necessary to modify the external tank thermal protection. Preliminary analysis indicates that tank separation can be performed with abort trajectories. Analyses also indicate that the abort capability of the OEHT shuttle is comparable to the baseline fully reusable vehicle and the system does not require use of an alternate landing site such as that required by the baseline vehicle for staging between 180 and 203 seconds.

Table 4-3. Orbiter External Hydrogen Tank Recovery Comparison

Separation time-sec	Baseline Fully Reusable Shuttle				OEHT Shuttle					
	Traj	Loads	Low q Control	High q Control	Traj.	Loads	Low q Control	High q Control	Tank Separation	Tank Disposal
$T_o + 20$	✓	✓	OK	OK	✓	✓	OK	OK	OK	Left or right turn for disposal in ocean area; all azimuth
$T_o + 80$	✓	✓	OK	OK	✓	✓	OK	OK	OK	
$T_o + 169$	--	--	--	--	O.A. or ✓	✓	OK	OK	OK	
$T_o + 176$ (OEHT Staging)	--	--	--	--	O.A.	✓	OK	OK	OK	OK
$T_o + 180$	✓	✓	OK	OK						
Alternate Landing Site — $180 \leq$ to ≤ 203										
$T_o + 203$	O.A.	✓	OK	OK						
$T_o + 213$ (Reusable Staging)	O.A.	✓	OK	OK						

NOTE: Alternate landing site not required for OEHT configuration.

Traj ✓ — Trajectory simulation for recovery at launch site completed
O.A. — Once-around abort capability

Loads ✓ — Recovery can be performed within load and heating constraints. Local interference heating between body and LH_2 tanks not assessed for atmospheric flight.

Control OK — Assessment of trim margins indicates adequate control at low and high q regions

4.2 SELECTED ORBITER DEFINITION

This section of the report summarizes the configuration and subsystem design definitions for the orbiter selected at the completion of Phase 1.

4.2.1 ORBITER DESIGN

The orbiter size selected is optimized to achieve a minimum-cost shuttle system in which the orbiter uses external hydrogen tanks. The selected concept is illustrated in Figure 4-7; it has a gross liftoff-weight of 1,123,000 pounds, including the external tanks. The vehicle is similar in aerodynamic profile to the fully reusable two-engine orbiter. It therefore provides comparable hypersonic performance but has a slightly degraded subsonic lift-to-drag ratio because of the increased base area required by installation of the third rocket engine. The tank arrangement is revised relative to the fully reusable system to reflect the incorporation of the external tanks. With these, the LO_2 for the main propulsion system is repositioned to a two-cell integral tank, similar in design to the integral

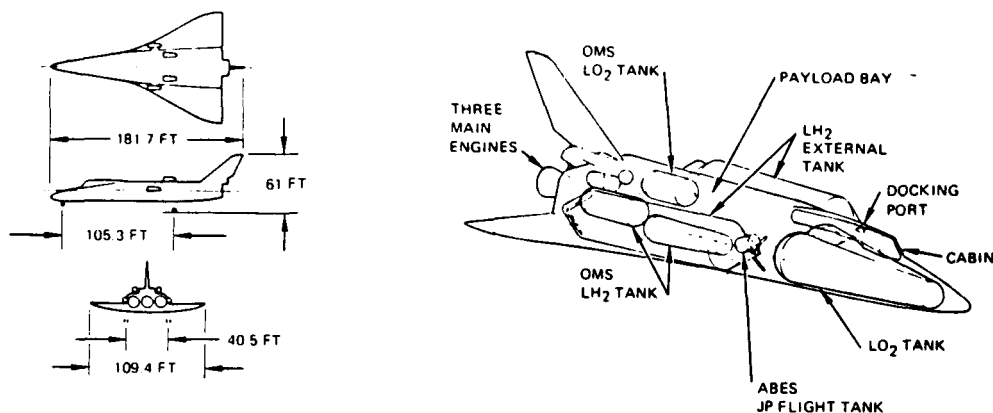| ITEM	EXTERNAL TANK	ITEM	EXTERNAL TANK
AERO/THERMAL	1,123 K	GROSS WEIGHT	1,123 K
ENTRY L/D	1.45	LH2 TANKS	2 EXTERNAL
SUB L D	6.25	LO2 TANKS	1 TWO CELL (FWD)
W/C S	77 LB/FT²	TUNNEL	144 IN
STRUCTURE TPS		MAIN ENGINES	3 (477 K EACH)
PRIMARY	TITANIUM	LH2 OMS TANK	2 (2 ON RH SIDE)
TANKS	ALUMINUM	LO2 OMS TANK	1 (ON LH SIDE)
HEAT SHIELD	REI	JP FUEL TANK	RH SIDE
PROPUL SYSTEMS		BODY REF LENGTH	1940 IN
MAIN ENGINES	3 AT 477K LB EACH	WING AREA	6763 FT²
ACPS THRUSTERS	29 AT 2100 LB EACH		
OMS THRUSTERS	3 AT 10,000 LB EACH		
ACPS OMS TANKAGE	INTEGRATED		
ABES	4 JTF22A 4 JP FUEL		

Figure 4-7. External Tank Orbiter

LH$_2$ tank for the fully reusable vehicle, to the nose of the orbiter. This results in space becoming available in the mid-body section for storage of the orbit maneuvering system (OMS) propellant, which is stored in two LH$_2$ tanks located on each side of the orbiter. This arrangement, selected because it resulted in the lowest weight system, results in a lateral offset of 3 inches in the vehicle center of gravity; this does not adversely affect vehicle control. The mid-body section is also used to accommodate fuel for the air-breathing engines in a JP fuel tank on the orbiter's right side. The width of the vehicle is increased to accommodate three main engines mounted horizontally across the vehicle. This arrangement was selected after examining a number of configurations, including a staggered main engine system. The arrangement of the OMS engines is similar to that of the fully reusable system and has three 10,000-pound-thrust engines located above the main rocket engine system.

The auxiliary control propulsion system (ACPS) for this orbiter consists of 29 thrusters, each delivering 2100 pounds of thrust. The propellant for these thrusters is located in the same tanks as the OMS propellant.

The total tank arrangement for the orbiter results in a high fineness ratio body with comparable aerodynamic characteristics to the fully reusable system. Aerodynamic surfaces are similar to those for the fully reusable vehicle with a slightly larger delta wing area to provide comparable landing speeds and reentry lift loading. Go-around capability for the space station logistics mission is achieved through incorporation of four JTF 22A-4 air-breathing engines using JP fuel.

The system changes developed for the external tank 0500 orbiter as compared with the baseline 161C orbiter are summarized in Table 4-4; they will be discussed in the various sections of this volume. Changes were required to the structure and Thermal Protection Systems (TPS) as well as to the main and auxiliary propulsion systems. The addition of the third main rocket engine required the addition of actuators in the hydraulic system. Checkout, monitoring, status display, and software changes were required for the avionics system. There are no changes to the air-breathing engine system, the ECLSS, the electrical and electrical power distribution, and control systems.

The basic structure and TPS concepts for the external tank system are similar to those for the fully reusable vehicle. The primary structure is titanium while the tanks are of aluminum.

Thermal protection is provided through the use of external reusable insulation except in those areas where temperatures exceed 2500 degrees; a carbon-carbon material is used there.

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Table 4-4. System Changes External-Tank Orbiter

System	Changes
Structure/Thermal Protection System	
• Structure	• External LH2 tanks • Integral LO2 tank • Configuration differences
• TPS	• Add TPS in tank interference heating area
Propulsion	
• Main propulsion	• Add third engine • Revise thrust levels • New tank locations • Deleted PU system • Revise propellant requirements
• Auxiliary propulsion	• Single LO2 tank } Revised location • Different LH2 tanks • Revised propellant requirements
• Air-breathing engines	• No change
Power	
• Electrical/mechanical power generation	• No change
• Hydraulic	• Add pitch/yaw actuators for third engine • Revised requirements
• Electrical power distribution/control	• No change
ECLSS	• No change
Avionics	• Checkout • Monitoring } Additions for • External tank • Third engine • Status display • Software

4.2.2 ORBITER AERODYNAMICS

The major factor in the orbiter design has been the shaping of the vehicle to achieve adequate aerodynamic performance. The major aerodynamic constraints imposed on the system are:

The requirement to provide approximately 1200-nautical-mile cross range capability

Requirement to provide pitch trim capability, and fly a programmed entry profile which will minimize TPS requirements

The need to minimize control system propellant requirements.

Figure 4-8 shows the hypersonic performance characteristics of the external tank orbiter concept. The vehicle lift coefficient and lift-to-drag (L/D) ratio are presented as a function of vehicle angle-of-attack. As illustrated, the maximum hypersonic L/D of the external hydrogen tank is approximately 2.2; for the desired entry angle-of-attack of 32.5 degrees, this hypersonic L/D is 1.45. The vehicle also has a comparable lift loading $W/C_L S$ of 77 pounds per square foot to that for the baseline system. Trim capability of the vehicle at hypersonic speeds is also shown and indicates that for the projected center of gravity range of 67 to 69 percent, the vehicle can be trimmed with elevon settings from -45 to +15 degrees.

The lateral/directional stability characteristics of the external tank orbiter at hypersonic speeds were also established. The vehicle is directionally unstable at angles of attack less than 45 degrees and this requires use of ACPS for directional control on entry. Subsonic aerodynamic characteristics of the external hydrogen tank orbiter are presented in Figure 4-9. As illustrated, the maximum subsonic lift-to-drag ratio is 6.25 achieved at an angle of attack of 10 degrees. Also illustrated for the vehicle is a neutral stability constraint which indicates that for the projected center of gravity range of 67 to 69 percent, the vehicle is stable up to an angle of attack of 28 degrees.

4.2.3 ORBITER TRAJECTORIES

Studies performed on the baseline system were directed towards establishing an orbiter entry trajectory that would minimize TPS requirements while satisfying the 1200-n.mi cross range requirement. These studies show that comparable system weights would be achieved with both constant angle-of-attack entry and entry profiles using the orbiter high angle-of-attack entry followed by transition down to a lower angle of attack. The entry profile is presented in Figure 4-10.

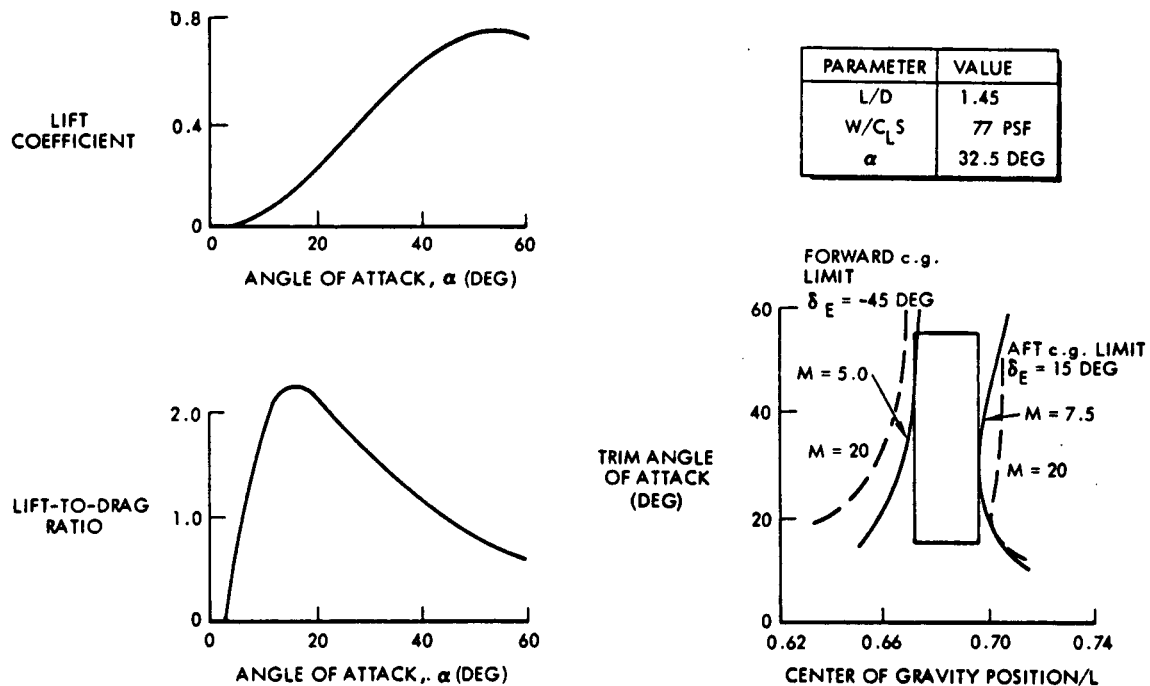| PARAMETER	VALUE
L/D	1.45
$W/C_L S$	77 PSF
α	32.5 DEG

Figure 4-8. Hypersonic Characteristics

This entry profile is based on a 32.5-degree angle-of-attack entry, which is maintained down to speeds of 3000 fps relative velocity. With this entry profile, the vehicle bank is modulated, starting at pull-up at 259,000 feet and terminated at 120,000 feet when angle-of-attack modulation is initiated. Following the vehicle entry, landing is achieved with an approach speed of 169 knots.

4.2.4 ORBITER CONTROL

Orbiter entry control encompassed the vehicle dynamic entry simulation to establish aerodynamic surface and ACPS control requirements. The conditions established to be critical for the aerodynamic surface are:

Elevon - Critical condition entry

Rudder - Critical condition approach for landing

Drag Brake - Critical condition approach for landing

In order to establish aerodynamic surface control requirements, simulations were performed for a bank reversal during entry and vehicle approach to touchdown under cross range conditions. The aerodynamic surface hinge moments rates, deflections, and duty cycle for these vehicle
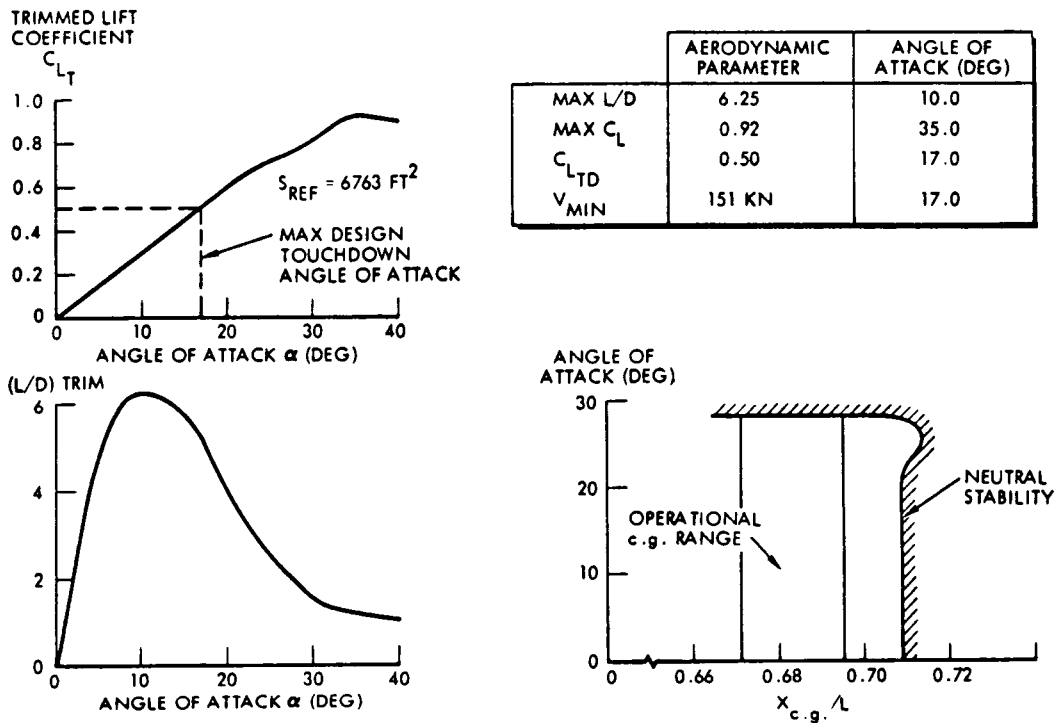
- 51 -

Figure 4-9. Subsonic Characteristics

	AERODYNAMIC PARAMETER	ANGLE OF ATTACK (DEG)
MAX L/D	6.25	10.0
MAX C_L	0.92	35.0
$C_{L_{TD}}$	0.50	17.0
V_{MIN}	151 KN	17.0

| MISSION EVENT	h (FT)	V_{REL} (FPS)	γ_{REL} (DEG)	α (DEG)	ϕ (BANK) (DEG)	$\bar{q}$ (PSF)	NZ	t (SEC)
① ENTRY INTERFACE	400,000	25,625	-0.86	32.5	0	0	0	0
② PULLOUT, START φMODULATION	259,250	25,411	-0.05	32.5	0→80	15	0.23	395
③ MAX LOAD FACTOR	182,650	12,572	-0.43	32.5	47	79	1.25	1,584
④ END φMODULATION	166,100	8,426	-0.49	32.5	15	65	1.05	1,782
⑤ START α MODULATION	120,850	3,000	-4.52	32.5	15→0	56	1.12	2,072
⑥ TERMINAL APPROACH INTERFACE	50,000	668	-7.59	15.0	0	85	1.00	2,447
⑦ LANDING	SEA LEVEL	169 KNOTS	-0.50	13.0	0	97	1.00	3,000
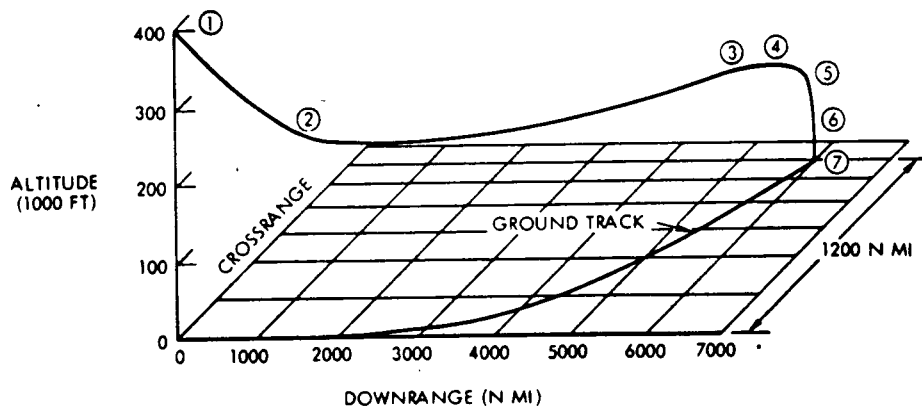
Figure 4-10. Orbiter Entry Mode

maneuvers are presented in Table 4-5. As indicated by the table, the maximum hinge moment is achieved on the elevon during entry with a requirement for 2.4 M inch pounds. The hinge moment requirement on the rudder is associated with an approach to landing condition with a requirement for 0.8 M pounds. Critical aerodynamic surface rates are experienced on approach and landing. With the requirements specified in the subject table, the hydraulic system actuators and power supply for the baseline reusable system are adequate for the external tank concept.

Analyses of the auxiliary control propulsion requirements during entry of the external tank orbiter are based on use of the same ACPS jet thrust as that used for the baseline system. The analysis also reflected the use of 29 thrusters located in similar positions to those for the fully reusable orbiter. Figure 4-11 shows the location of these ACPS jets and the rate requirements to be achieved with this system to provide adequate control authority following two failures. As indicated, the specified ACPS thrusters and their associated locations provide adequate control authority for the external tank concept.

4.2.5 ORBITER LOADS

The orbiter design loads are presented in Section 4.1.5.

4.2.6 ORBITER THERMAL ENVIRONMENT

Thermal analyses were conducted to establish the design thermal environment and thermal protection system requirements considering both the ascent and reentry flight conditions. These data were used in the TPS design studies and orbiter weight analysis.

The major difference between the OEHT configuration and the reusable orbiter is the presence of the external tank and the resulting interference heating between the tank and orbiter fuselage.

The results of the thermal analysis are presented in Figure 4-12, which shows typical temperature on the orbiter and the 2200 F and 850 F isotherms. The materials, RPP, REI, and titanium, required for each temperature zone are shown. The majority of the orbiter TPS is required for entry flight during the maximum crossrange maneuver. However, as indicated on the figure (cross-hatch area), the interference heating between the tanks and orbiter requires approximately 2900 square feet of additional area to be covered with REI.

A study of a tank fairing to reduce the orbiter TPS weight penalty was conducted, and the results are summarized in Section 4.3.2.

Table 4-5. Orbiter Aerodynamic Surface Requirements

Deflections in Degrees Rates Deg Sec Moments 10^6 in lb		Entry Maneuver Bank Reversal	Cruise Approach Touchdown Crosswind	Duty Cycle -RMS-	Max Hinge Moment At Zero Rate	Max Rate At Zero Hinge Moment	Max Deflection
Elevon	Deflection	-20	-11	0.6			+15 to -45
	Rate	7.5	15	3.0		±20	
	Hinge Moment	2.4	0.81	0.42	3.7×10^6 (2.4×10^6)		
Rudder	Deflection	5	3	±0.4			±10
	Rate	7.5	8	3.0		±12	
	Hinge Moment	0.17	0.18	0.02	0.34×10^6 (0.25×10^6)		
Drag Brake	Deflection	40	30	-			0 to 35
	Rate	1.0	4	-		±7	
	Hinge Moment	0.25	0.8	-	1.2×10^6 (0.91×10^6)		

Surface Freq Response: (1) Elevons and Rudders 4 CPS $\zeta=0.6$
                       (2) Drag Brakes 5 sec first order lag
                       (3) Values in brackets: fail safe requirements

- Requirements verified by Control Simulations

- Conditions Critical to requirements:

  Elevon - Entry
  Rudder - Approach for landing
  Drag Brake - Approach for landing

- Fully Reusable Baseline System Satisfies requirements

- 2100 LB THRUST ACPS JETS SATISFACTORY

- Wp REQUIREMENT IS 4697 LB, COMPARABLE TO FULLY REUSABLE BASELINE
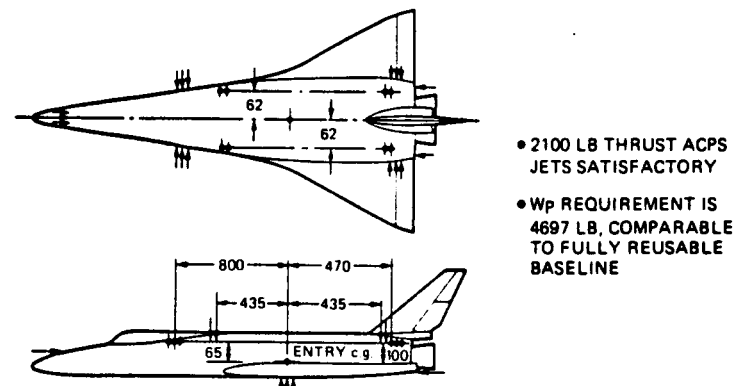
SINGLE ENGINE THRUST LB	ENGINES PER AXIS	AXIS	LEVER ARM FT	NOMINAL ACCELERATION	FAIL SAFE ACCELERATION	REMARKS
2100	6	YAW	66	1.87 ^{o}/SEC^2	1.16 ^{o}/SEC^2	CRITICAL DURING REENTRY (SIZES ACPS THRUST LEVEL)
2100	4	PITCH	36	0.9 ^{o}/SEC^2	0.5 ^{o}/SEC^2	SIZED TO SATISFY ON-ORBIT HANDLING QUALITIES
2100	4	ROLL	5	1.0 ^{o}/SEC^2	0.5 ^{o}/SEC^2	SAME AS PITCH

Figure 4-11. Orbiter ACPS Requirements

4.2.7 STRUCTURE AND TPS

As previously indicated, the structure and TPS concepts for the external hydrogen tank orbiter are identical to those for the fully reusable orbiter vehicle. The structural arrangement is illustrated in Figure 4-13. The nose of the external tank orbiter is a skin stringer frame construction using titanium. The forebody section of the vehicle accommodates an integral load-carrying LO_2 tank fabricated from aluminum with external stiffeners and standoff frames for the external insulation. Loads from this tank are transmitted to the fuselage primary load-carrying structure through a titanium skin-stringer frame skirt. This forward LO_2 tank design results in modification of the load paths relative to the fully reusable vehicle.

In the mid-body section, main LO_2 propellant tanks are eliminated, and the primary load carrying shell is a skin-stringer frame construction. The internal titanium truss structure, used on the fully reusable vehicle to react LO_2 tank axial loads, is maintained in the external tank system to transmit swing torsion loads to bulkheads forward and aft of the cargo bay. The wing and other elements of the mid-body structure section are similar to those structures for the reusable baseline shuttle system.

The aft-body section of the external tank orbiter is revised from the reusable baseline to accommodate the third main rocket engine. The engine
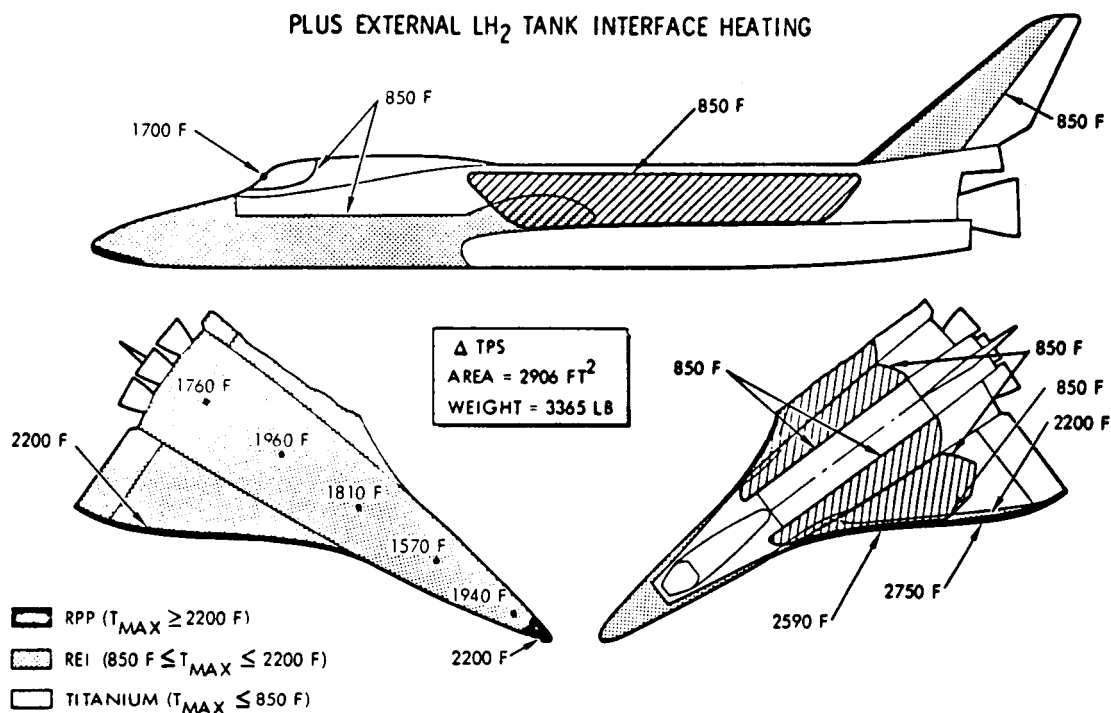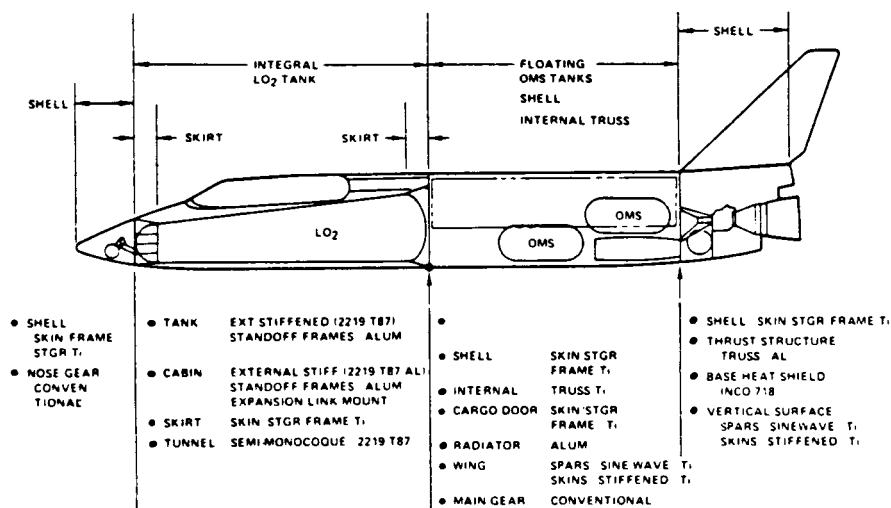
- 55 -

RPP ($T_{MAX} \geq 2200$ F)

REI (850 F $\leq T_{MAX} \leq 2200$ F)

TITANIUM ($T_{MAX} \leq 850$ F)

Δ TPS
AREA = 2906 FT^2
WEIGHT = 3365 LB

Figure 4-12. Orbiter TPS Requirements

ITEM	EXTERNAL TANK
FWD BODY	INTEGRAL LO$_2$ TANK
MID BODY	—
INTERNAL TRUSS	—
THRUST STRUCTURE	3 ENGINE
EXTERNAL LH$_2$ TANK	ADDITIONAL LOAD PATH

Figure 4-13. Structural Design and Materials Comparison

SD 71-141-1

thrust structure is modified to incorporate additional diagonal members and three shear panels. Due to the relocation of the thrust structure, the vertical tail intersects forward of the main thrust beam. The aft-body section is also revised to transmit external hydrogen tank axial forces through a drag member to the main engine thrust structure.

The thermal protection system concept for the external tank orbiter is identical to that for the fully reusable baseline. The area of the vehicle which requires external insulation is, however, increased to accommodate interference heating which occurs between tank and orbiter during the boost phase of the mission. The TPS concept and weights associated with the thermal protection system are illustrated on Figure 4-14 which indicates that TPS weights for the external tank orbiter are approximately 1200 pounds greater than the weights for the fully reusable orbiter. This thermal protection system is based on the use of external tanks with a 60-degree nose cone. A trade study was performed to assess the merit of a fairing between tank and orbiter fuselage and wing. This trade study, which is documented in Section 4.3.7, indicated that the TPS weight reduction associated with the orbiter-to-tank fairing would be more than offset by the increased weight of the fairing. The increased cost of the fairing also indicates that it is more desirable to provide additional external insulation on the orbiter for protection against interference heating effects during the boost phase of the mission.
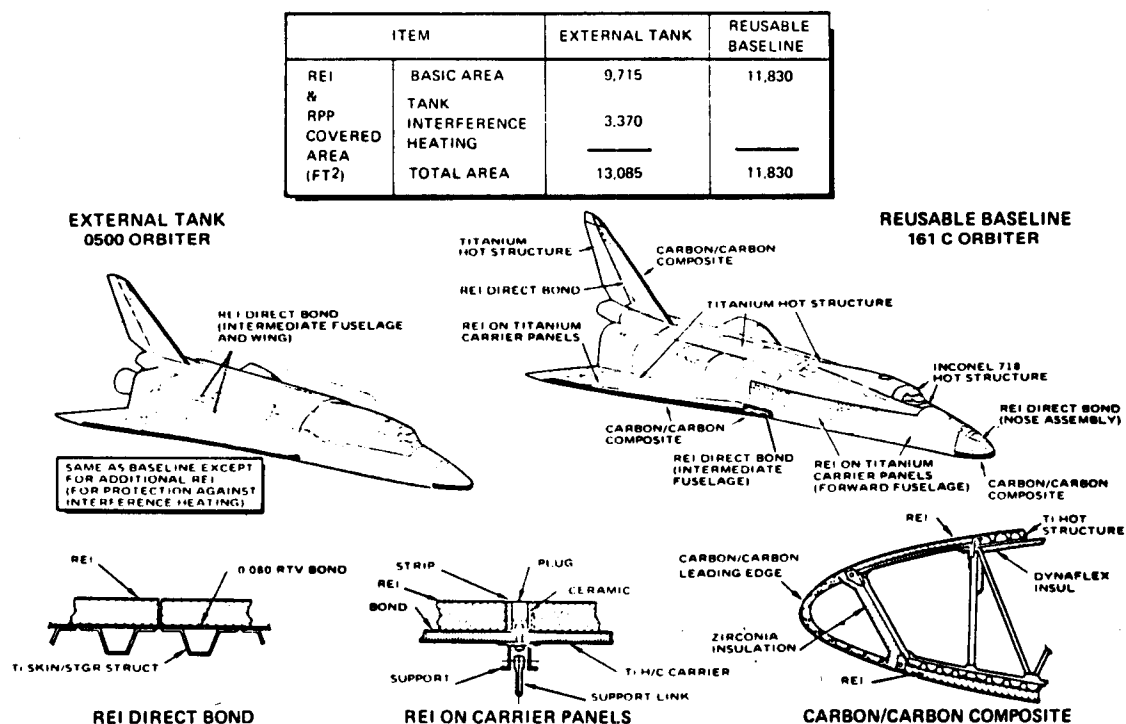
ITEM		EXTERNAL TANK	REUSABLE BASELINE
REI & RPP COVERED AREA (FT2)	BASIC AREA	9,715	11,830
	TANK INTERFERENCE HEATING	3,370	—
	TOTAL AREA	13,085	11,830

Figure 4-14. Orbiter TPS Comparison

4.2.8 PROPULSION AND FLUID SYSTEMS

The most significant change to the fully reusable orbiter propulsion systems for the external hydrogen tank concept is the use of three main rocket engines compared to two engines on the baseline system. These three engines each provide 477 pounds of vacuum thrust and utilize the same expansion ratio (ϵ = 150:1) as the engines on the baseline system. Propellant is delivered to these main rocket engines from the single two-cell LO_2 tank in the orbiter and the two cylindrical external hydrogen tanks. A trade study was performed for the external tank orbiter concept to assess the merit of a propellant utilization (PU) system. Due to the orbiter cost associated with incorporation of a PU system in the external hydrogen tanks, it was concluded that for this concept, an LH_2 propellant bias results in a more cost-effective system. External tanks on the orbiter therefore utilize point sensors for propellant gauging only, and the capacitance probes used for propellant utilization on the reusable system are precluded from the external orbiter tank concept. A resulting weight penalty was therefore incurred in the LH_2 concept to allow for the 1200 LH_2 propellant bias. Main propulsion system interfaces between external LH_2 tanks and orbiter include a vent system, main propellant feed line, and propellant recirculation. These interfaces utilize a self-sealing valve on the external tank side of the interface and mechanically driven shutoff valves on the orbiter side of the interface. A trade study was performed to determine the relative merit of single versus dual vent systems for the external hydrogen tank system. The trade study indicated that a lower cost program could be achieved through the use of a single vent on the orbiter for the two external hydrogen tank system. The schematic for the main propulsion system is presented in Figure 4-15.

A number of options were considered for the installation of the three main rocket engines. Options considered during Phase 1 of the program included mounting of the three engines horizontally and a staggered engine arrangement resulting in a deep orbiter body. Phase 1 studies resulted in selection of the engine arrangement with three engines mounted horizontally to obtain the best aerodynamic characteristics.

During Phase 2 of the study, different engine installations were considered to establish the installation which would result in the minimum base area on the orbiter while providing adequate control authority following failure of one of the main engines. As illustrated in Figure 4-16, an engine arrangement which provides for gimbal capability on all three engines and for all engines thrusting parallel to the vehicle centerline under normal conditions results in the smallest vehicle base area. This concept identified in Figure 4-16 was therefore selected for the baseline external tank orbiter.
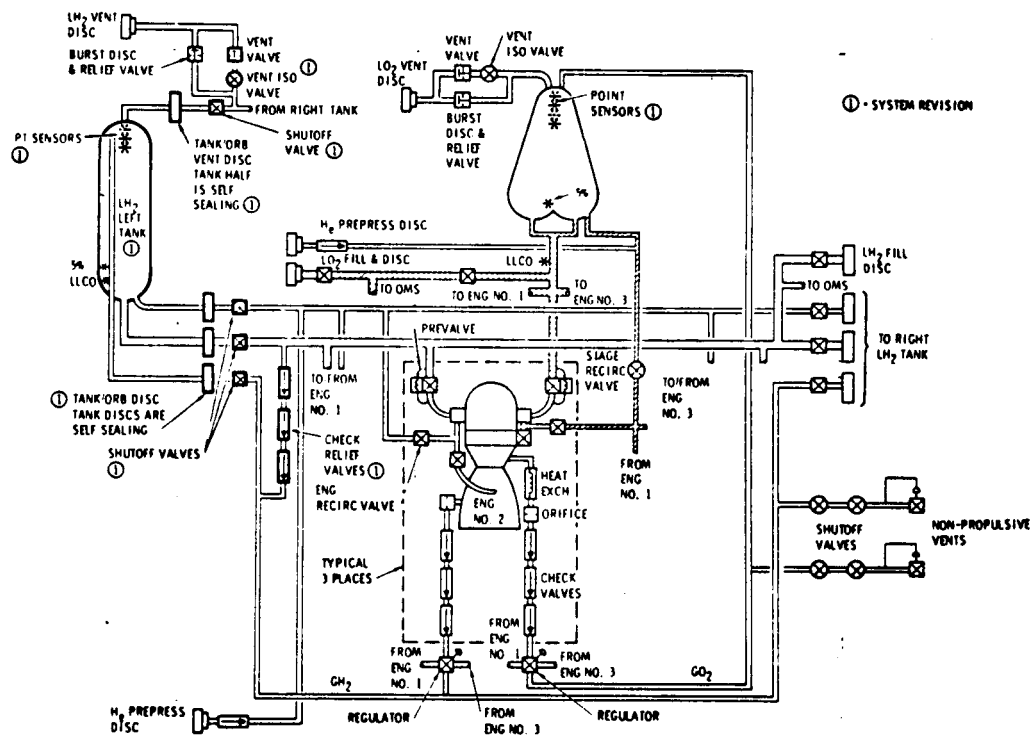
Figure 4-15. Main Propulsion System

Figure 4-16. Three Main Rocket Engine Yaw
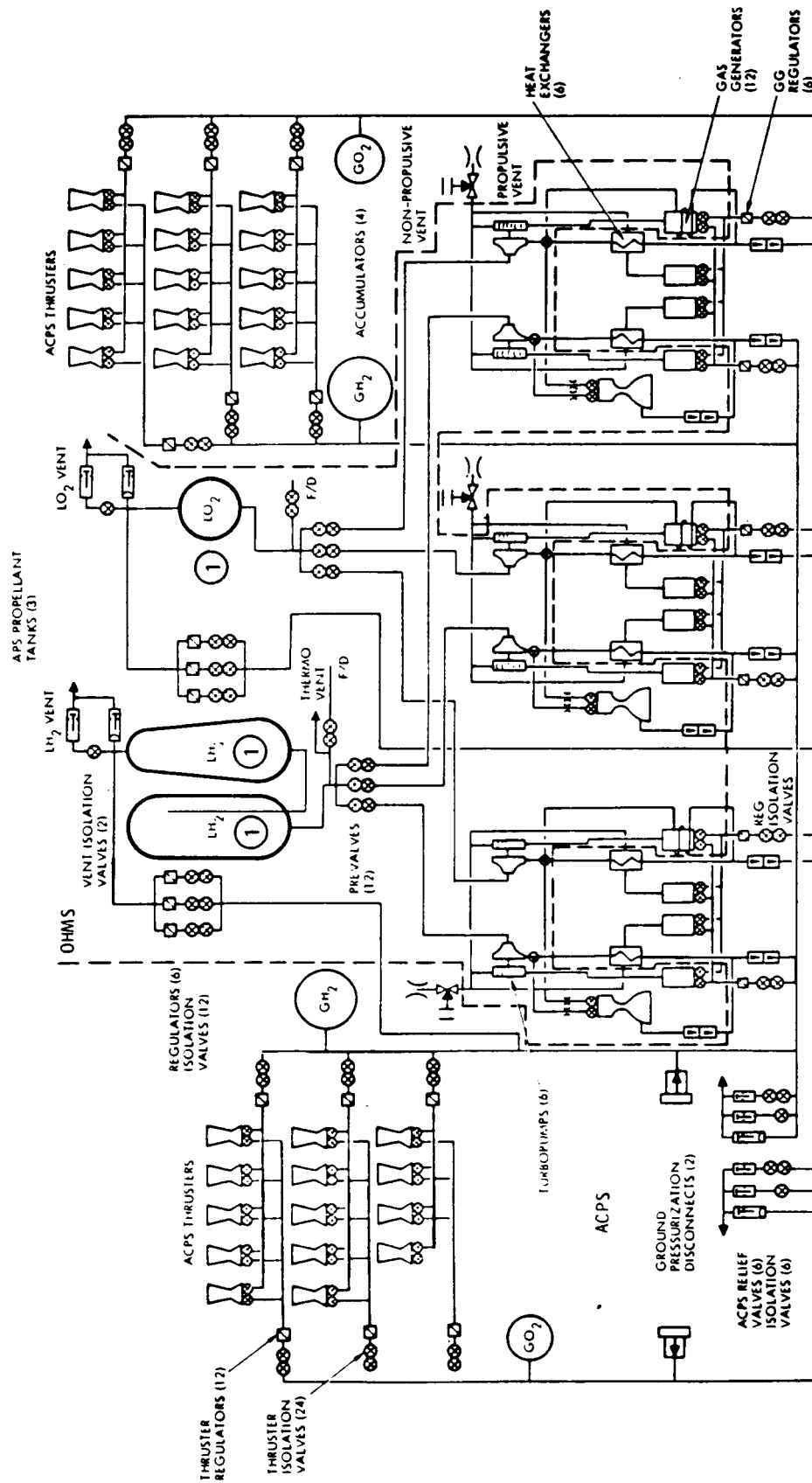Control Trade Study

The orbiter auxiliary propulsion system performs the dual functions of providing for orbit maneuvers and attitude control. The orbit maneuvering propulsion system uses three LO_2/LH_2 10,000-pound thrust engines. The attitude control system uses 29 GO_2/GH_2 thrusters, each with a vacuum thrust of 2100 pounds. Propellant for the orbit maneuvering and attitude control systems is contained in high performance dewar type tanks, employing an LH_2 regenerative vent subsystem to minimize ullage venting and boiloff. Two LH_2 tanks and a single LO_2 tank are provided to accommodate the propellant for the two previously described subsystems. The significant change from the fully reusable orbiter is the use of a single LO_2 tank as opposed to the dual tanks for the fully reusable baseline system. Ullage pressurization with GO_2 and GH_2 before and during operation of the attitude control system is extracted from gaseous storage accumulators. Orbit maneuvering system tapoff and the gas generator heat exchanger provide GH_2 and GO_2 pressure, respectively, during the orbit maneuvering mode. Three gas-generator-driven LO_2 and LH_2 turbo pump sets are common to both of the previously described subsystems. The integrated schematic for the ACPS and the orbit maneuvering system is presented in Figure 4-17.

The air breathing engine system on the external tank orbiter uses four JTF 22A-4 engines which are identical to those proposed for the fully reusable orbiter. This concept is therefore identical for the two orbiter concepts.

The hydraulic system for the external tank orbiter is identical to that used for the fully reusable orbiter except for the provision of a hydraulic supply and actuators to gimbal the additional main engine.

Due to the removal of the LH_2 propellant tanks from the inside of the orbiter fuselage, purging requirements for the external tank orbiter are revised from those for the fully reusable system. A comparison of the purge requirements for the LH_2 is presented in Figure 4-18. As illustrated by the figure, the only significant design change for the external tank orbiter is associated with elimination of the inerting gas requirements for the LH_2 tanks, which is normally required after landing for the fully reusable system. Due to changes in volume also, the purge gas requirement for the external tank orbiter is reduced from 7090 cubic feet for the fully reusable system to 4090 cubic feet for the orbiter hydrogen concept.

The ECLSS design for the OEHT concept is identical to that for the fully reusable orbiter.

Figure 4-17. Orbiter Integrated ACPS/OMS

- MAIN PROPELLANT TANKS
- OMS PROPELLANT TANKS
- CAVITIES

PURGE GAS	PRELAUNCH PURGING			POST LANDING SAFING			GROUND MAINTENANCE PROTECTION		
	LH$_2$ TANKS	LO$_2$ TANKS	CAVITIES	LH$_2$ TANKS	LO$_2$ TANKS	CAVITIES	LH$_2$ TANKS	LO$_2$ TANKS	CAVITIES
DEHUMIDIFYING AIR	✓	✓					✓	✓	✓
DRY NITROGEN	✓①		✓	⊗②	✓③	✓			
GAS (H$_2$ OR O$_2$)	✓								
LIQUID (H$_2$ OR O$_2$)	✓	✓							

DEHUMIDIFYING LEVEL DEW POINT 65 F
① SAFE O$_2$ LEVEL 1% BY VOLUME
② SAFE H$_2$ LEVEL 1% BY VOLUME
③ SAFE O$_2$ LEVEL 20% BY VOLUME
⊗ DELETED FOR EXTERNAL LH$_2$ TANK ONLY

EXTERNAL TANK TO BASELINE ORBITER DIFFERENCES
- EXTERNAL LH$_2$ TANKS
- LO$_2$ TANK FWD
- SMALLER VEHICLE
- LARGER TANKS
- EMPTY MID BODY

FLOW RATES (PRELAUNCH)
161C BASELINE ORBITER
- ~ 44,000 LB/HR
0500 EXTERNAL TANK ORBITER
- ~ 35,000 LB/HR

EXCLUDES CARGO BAY
- FLOW RATE DEPENDS ON CARGO

Figure 4-18. Purge Requirements Comparison, External Tank Orbiter

4.2.9 AVIONICS

The avionics concept for the external tank orbiter is identical to that used for the fully reusable system. Design changes are associated with the displays and command equipment for jettison of the external hydrogen tanks. The instrumentation previously required for the propellant utilization system on the reusable system is also eliminated.

4.2.10 MASS PROPERTIES

Initial orbiter weights based on the vehicle synthesis performed during Phase 1 of the program are presented in Table 4-6. These vehicle subsystem weights were updated as a result of subsystem design analyses. The updated weights are also presented in Table 4-6.

Table 4-6. Orbiter Launch Summary Weight Statement

CODE	SYSTEM	INITIAL WEIGHT			REVISED WEIGHT		
	CONFIGURATION MODEL 0500 ORBITER – EXTERNAL TANK		BY NR		DATE		
1.0	WING GROUP	17670			25200		
2.0	TAIL GROUP	4458			3432		
3.0	BODY GROUP	54843			51811		
4.0	INDUCED ENVIR PROTECT	40487			39290		
5.0	LANDING, DOCKING	14917			17514		
6.0	PROPULSION, ASCENT	29205			30169		
7.0	PROPULSION, CRUISE	–			544		
8.0	PROPULSION, AUXILIARY	13294			14360		
9.0	PRIME POWER	3892			2717		
10.0	ELECTRICAL CONV & DIST	2954			3292		
11.0	HYDRAULIC CONV & DIST	1184			1143		
12.0	SURFACE CONTROLS	1895			2021		
13.0	AVIONICS	3559			3789		
14.0	ENVIRONMENTAL CONTROL	3282			3624		
15.0	PERSONNEL PROVISION	984			989		
16.0	RANGE SAFETY	16400			18121		
17.0	BALLAST						
18.0	GROWTH						
19.0							
	SUBTOTAL (DRY WT)	209024			218016		
20.0	PERSONNEL	614			618		
21.0	CARGO	40000			40000		
22.0	ORDNANCE	4768			4107		
23.0	RESIDUAL FLUIDS						
24.0							
	SUBTOTAL (INERT WT)	254406			262741		
25.0	RESERVE FLUIDS	6960			5684		
26.0	INFLIGHT LOSSES	6026			6779		
27.0	PROPELLANT-ASCENT	677363			677363		
28.0	PROPELLANT-CRUISE	–			–		
29.0	PROPELLANT-MANEUV/ACS	35111			35111		
30.0							
	TOTAL WEIGHT · LB	979866			987678		

DESIGNATIONS:

   ITEM
    A
    B
    C
    D
    E
    F

NOTES & SKETCHES

4.3 EXTERNAL ORBITER TANKS

The evaluation of the external orbiter tanks through the Phase 1 portion of the study and at the start of the Phase 2 portion of the study is shown on Figure 4-19. As noted, the cylindrical tanks with TPS protection for entry was analyzed during Phase 1 of the program. The tank structure was monocoque with cork ablator for entry protection. The tanks were located on the orbiter with a compression strut at the tank aft end, transmitting longitudinal loads to the orbiter engine thrust structure. Pyro actuators were selected for jettison tank capability and solid rocket motors were utilized for the roll and deorbit systems.

The tank definition at the end of Phase 1 was carried over into the start of Phase 2 except for one major revision: analysis indicated that fragmenting of the tank on entry is acceptable and it was therefore decided that no special entry protection would be utilized on the Phase 2 tank design. The analyses leading to the decision to allow tank breakup on entry is presented in Section 4.3.4.

4.3.1 SELECTED TANK CONFIGURATION

The tank configuration selected for design is defined on the orbiter configuration drawing, VB70-0500"B" change, which with all of the other design layouts is filed in the Appendix A of Volume II. A simplified picture of the tank configuration is given on Figure 4-20. This shows the cylindrical tank housing the propellant. A forward fairing is used to provide the aerodynamic shape for boost and to house equipment (deorbit motors, roll motors, etc.).

4.3.2 TANK DESIGN

The requirements for which the tank was designed were to house the LH$_2$ propellant and to provide installation for components of the propulsion, retro, and roll systems. The tank was designed for installation, separation, and final jettison from the sides of the orbiter. The tank structure and TPS provide the required structural and thermal integrity during the boost phase.

The preliminary design of the external tank is shown on drawing VB70-3100 which locates all of the system components in or on the tank assembly. A simplified presentation is given on Figure 4-21 which defines the propellant tank assembly, the aft support structure, and the aft propellant lines leading to the orbiter disconnect interface.
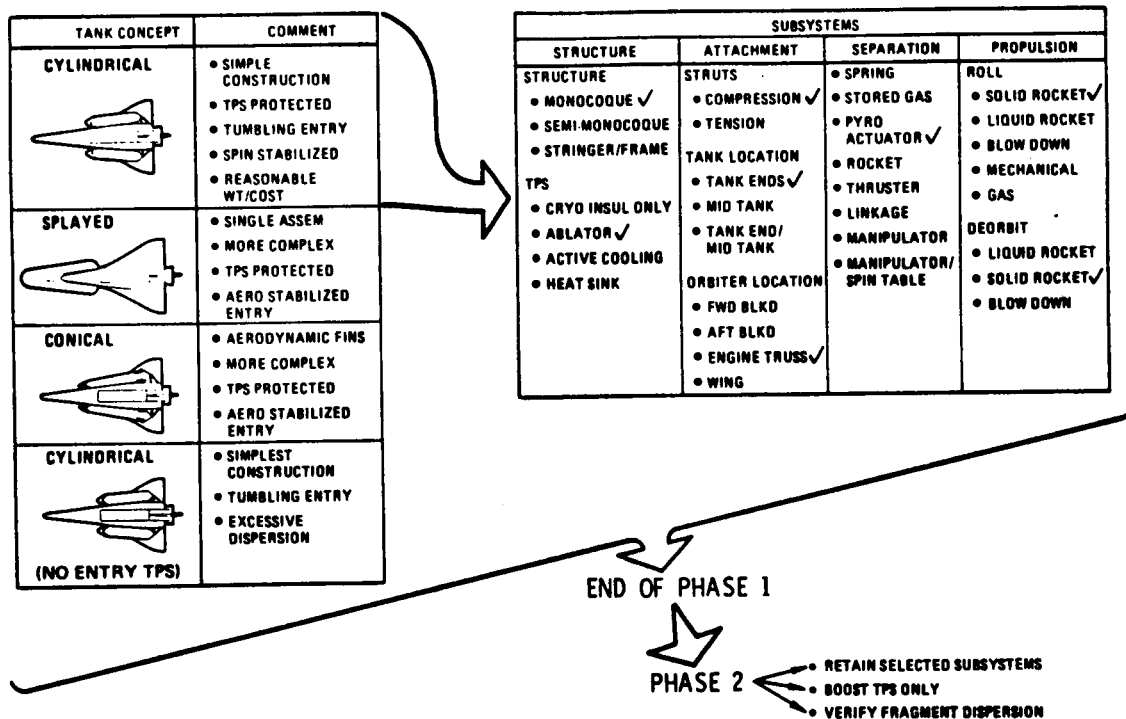
- 64 -
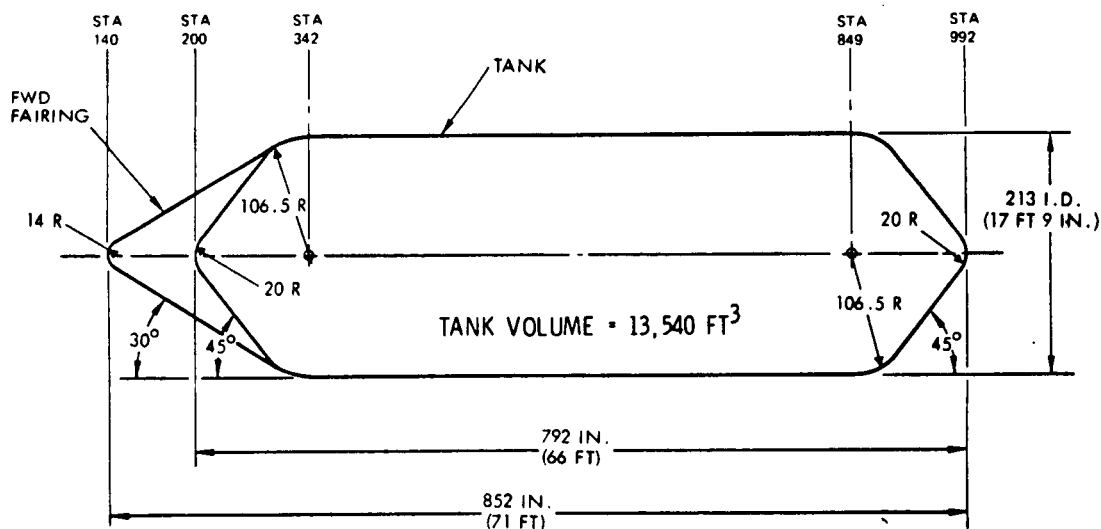
Figure 4-19. External LH$_2$ Tank Evolution

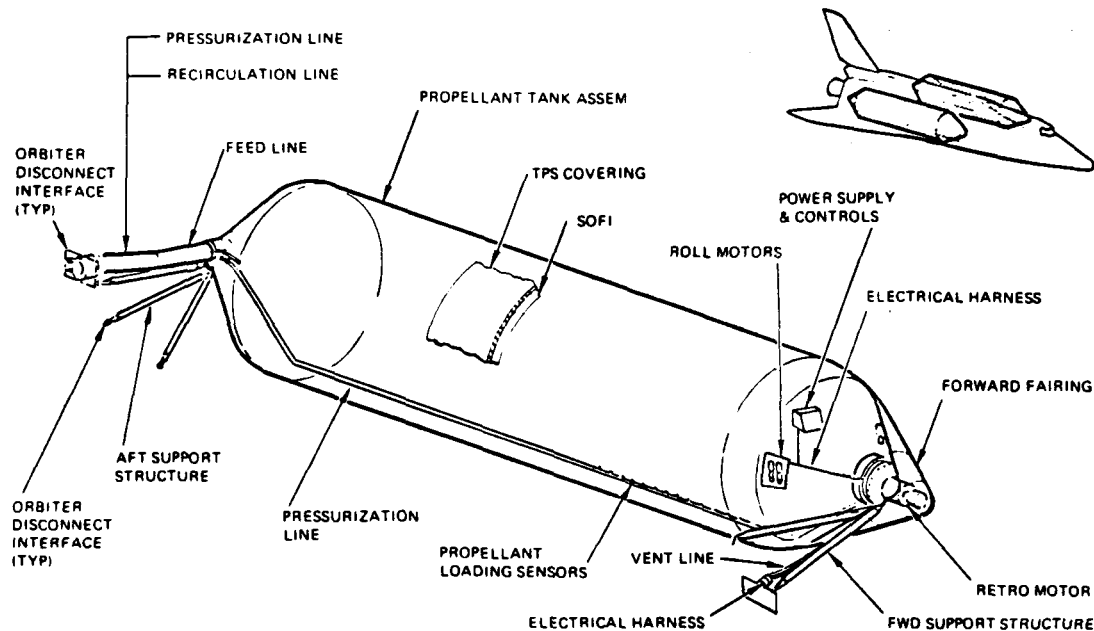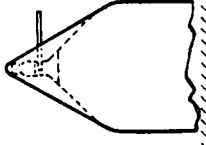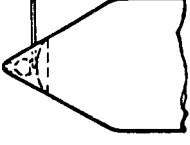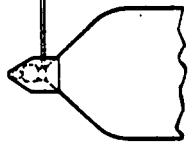Figure 4-20. LH$_2$ Tank Configuration

Figure 4-21. LH$_2$ Tank Assembly

The retro motor and the roll motors, as well as the power and control equipment, are installed in the forward fairing. A hinged forward support truss provides a stabilized determinate structural support for the tank. The vent lines and the electrical harnesses are mounted to one of the support struts--and also lead to the orbiter disconnect interface.

Various trade studies were performed during the preliminary design of the tank assembly. Figure 4-22 illustrates three options for the tank forward end shape. The tank configuration with a 45-degree cone-integral tank front end was rejected because of the high drag factors. The tank with a 60-degree conical-fairing and 90-degree forward bulkhead was selected on the basis of least weight and reasonable design factors.

A major trade study was performed to establish whether or not to use a side fairing between the external tank and the orbiter body and wing to minimize the boost temperature impingement effects. The tradeoff and results are shown on Figure 4-23. The configuration without a side fairing between orbiter and tank is subjected to interference heating which necessitates additional TPS for the orbiter. The use of a side fairing reduces the orbiter TPS requirement associated with interference heating but the weight of the fairing is increased. The net result is that the use of a side fairing results in comparable orbiter and tank weight to a system without the side fairing. The tank without a side fairing was therefore selected to allow the use of a simpler tank, prevent increased development costs for the additional fairing, and reduce the throw-away cost of the tank for each orbiter flight.

ITEM	60° CONE-FAIRING	60° CONE-INTEGRAL TANK	45° CONE-INTEGRAL TANK
CONFIG			
DRAG	LOW	LOW	HIGH (ΔGLOW · 25K)
TANK BULKHEAD	45° COMMON ENDS	60° DIFFERENT ENDS	45° COMMON ENDS
MANHOLE COVER	4.5 FT DIA COMMON ENDS	8 FT DIA DIFFERENT ENDS	4.5 FT DIA COMMON ENDS
FAIRING SIZE	LARGE	SMALL	COMPACT
FAIRING ATTACH	TANK RING	MANHOLE COVER RING	MANHOLE COVER RING
ΔMETAL WT (LB)	+152	0	—
ΔTPS WT (LB)	0	+198	—
TOTAL ΔWT (LB)	0	+ 46	—

● EXTERNAL SOFI   ● CORK TPS   ● PER TANK

Figure 4-22. External LH$_2$ Tank Forward End Trade Study

The disposal of the ejected tank assembly is based upon delaying breakup on entry by utilizing the inherent tank structure and TPS capability as designed for the external loads and heating applied during boost. The tank pressure prior to tank ejection was reduced to 5 psi to decrease the stress level in the tank skins, and thereby allow a greater temperature capability on the tank structure prior to tank failure during reentry. Figure 4-24 indicates the structural tank integrity through boost, tank jettison retro, and tank breakup. Tank breakup at approximately 350,000 feet is caused either by the reentry temperature buildup and/or the aerodynamic pressure buildup. As indicated in Section 4.3.4 tank failure at 350,000 feet is acceptable for the dispersion of the tank segments in the Indian Ocean.

4.3.3 TANK SEPARATION

Requirements imposed on the external hydrogen tank separation system are to provide an acceptable tank attitude to limit the impact dispersion on deorbit and also to achieve clearance between the tank and orbiter during the separation and deorbit operation. This section of the report defines the dynamic analyses associated with the separation and also provides a summary description of the selected system design.
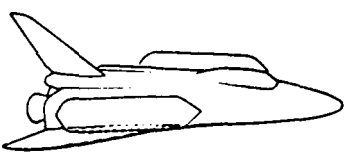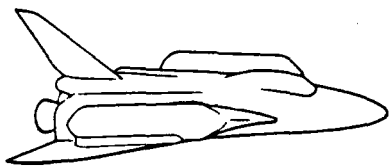
ITEM	NO SIDE FAIRING	SIDE FAIRING
CONFIG • WTS PER-2 TANKS		
BOOST THERMAL PERTUBATION	MAXIMUM	MINIMUM
ORBITER TPS (LB)	3365	100
TANK CONSIDERATIONS	HOUSE PROPELLANT	HOUSE PROPELLANT SUPPORT FAIRING LOADS
TANK CONSTRUCTION	MONOCOQUE	MONOCOQUE + FRAMES
FAIRING CONSIDERATIONS	—	REACT AIR LOADS ACCOMMODATE TANK GROWTH DEFLECTIONS
FAIRING CONSTRUCTION	—	ALUM SKIN/STRINGER/FRAME CORK TPS
TRUSS STRUCTURES PROPELLANT LINES	LOCAL FAIRINGS/TPS PROTECTION	PROTECTED BY SIDE FAIRING
TANK FAIRING WT (LB)	20,915	24,190
NET ORBITER Δ WEIGHT (LB)	24,280	24,290
DELTA PROGRAM COST	0	GREATER

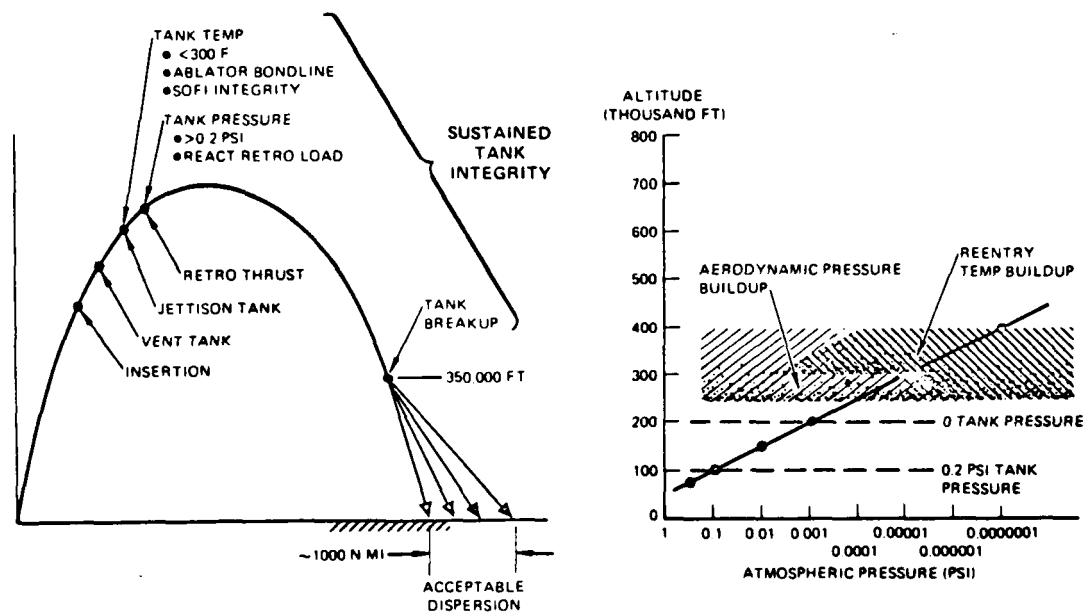Figure 4-23.   External LH$_2$ Tank Side Fairing Trade Study

Figure 4-24.   LH$_2$ Tank Disposal Implementation

4.3.3.1 Tank Separation Dynamics

The dynamic analysis for separation of the orbiter external hydrogen tank encompassed two separation mode options. The first option is based upon separation using the pyrotechnic actuation system; upon the achievement of adequate clearance between the tanks and the orbiter, the deorbit motors are to be fired. Option two is based upon separation of the tank from the orbiter and followed by spinning of the tank about its longitudinal axis to stabilize the tank and minimize attitude errors.

The tank separation analysis is based upon the tank being separated with a variation of 10 percent in the impulse provided by the two separation actuators and an uncertainty of one foot in the center-of-gravity location. These variations in tank and separation mechanism characteristics can result in attitude rates of approximately 3 degress per second on the tank while achieving a displacement velocity of 5 feet per second.

When the tank separation dynamics were analyzed without a spin stabilization system, variations in the separation system impulse were considered. Figure 4-25 defines the trajectory for the tank relative to the orbiter based on various deorbit motor ignition times following separation. Clearance distances and attitude errors between the tank and orbiter are also illustrated in this figure as a function of deorbit ignition time and variations in the tank separation impulse. The tank entry analysis reported in Section 4.3.4 indicates that attitude errors up to 30 degrees will result in acceptable tank dispersion. As illustrated by Figure 4-25, ignition of the deorbit motors approximately 8 seconds after separation will result in tank attitude error of approximately 30 degrees and will provide a clearance of approximately 150 inches where a 10 percent variation in separation actuator impulse is experienced. It is apparent, therefore, that where the variation in separation actuator impulse is within 10 percent, acceptable tank attitude errors and clearance can be achieved without a spin system. As indicated by the figure, however, a variation of 20 percent in the impulse provided by each separation actuator will reduce the clearance between tank and orbiter to approximately fifty inches. This clearance distance is considered marginal for safety. It is also noted that variations in tank center of gravity greater than one foot uncertainty specified on the figure will further degrade clearance distances. It was, therefore, concluded that the baseline external tank system would have a spin stabilization system. Follow on studies should continue to evaluate both separation system concepts.

The tank separation dynamics using a spin stabilization system are illustrated in Figure 4-26. This representative separation mode is based on the ignition of spin motors three seconds after tank separation and spin rates of approximately three radians per second about the tank longitudinal axis are achieved. Upon completion of tank spinning, the tank has a coning action
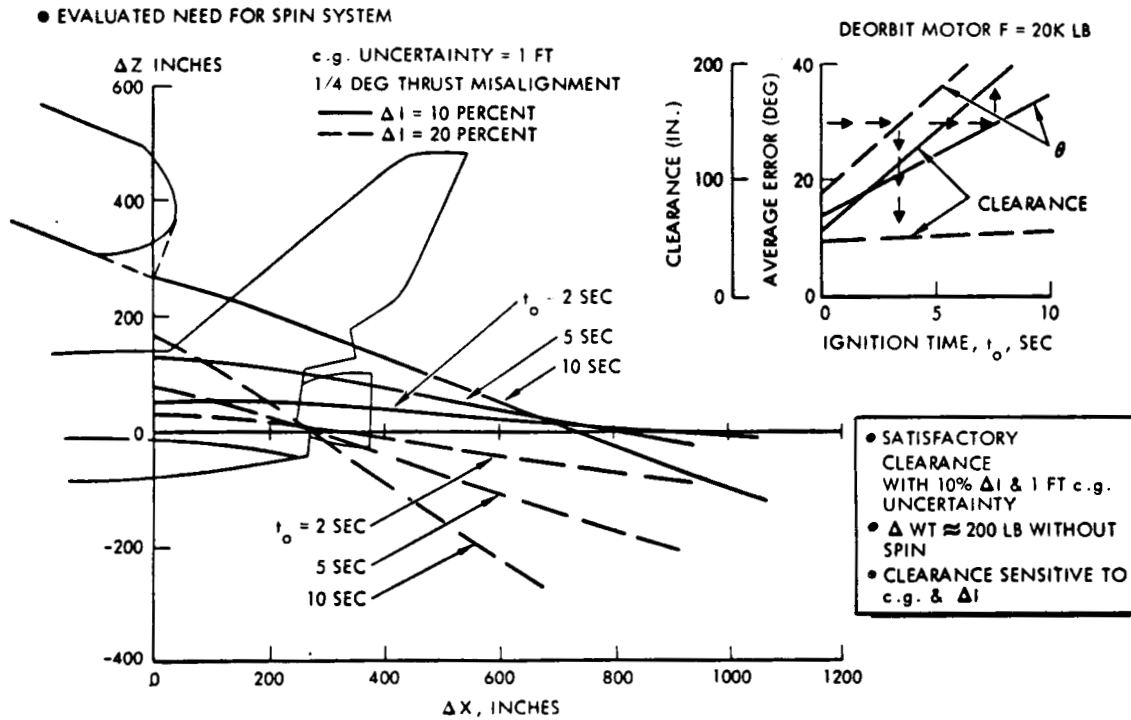
Figure 4-25. Tank Clearance During Retro (No Spin)

SEPARATION

FORCE = 20,000 LB
STROKE = 4 INCH
10% DIFF IMPULSE
CG UNCERT = 1 FT

FIRE SPIN MOTORS

Y = 15 FT
T = 3 SEC
THRUST = 4 X 550 LB
ΔT = 2.4 SEC
ω_0 = 2.9 RAD/SEC

END SPIN-UP

Y = 27 FT
T = 5.4 SEC
θ = 15 DEG

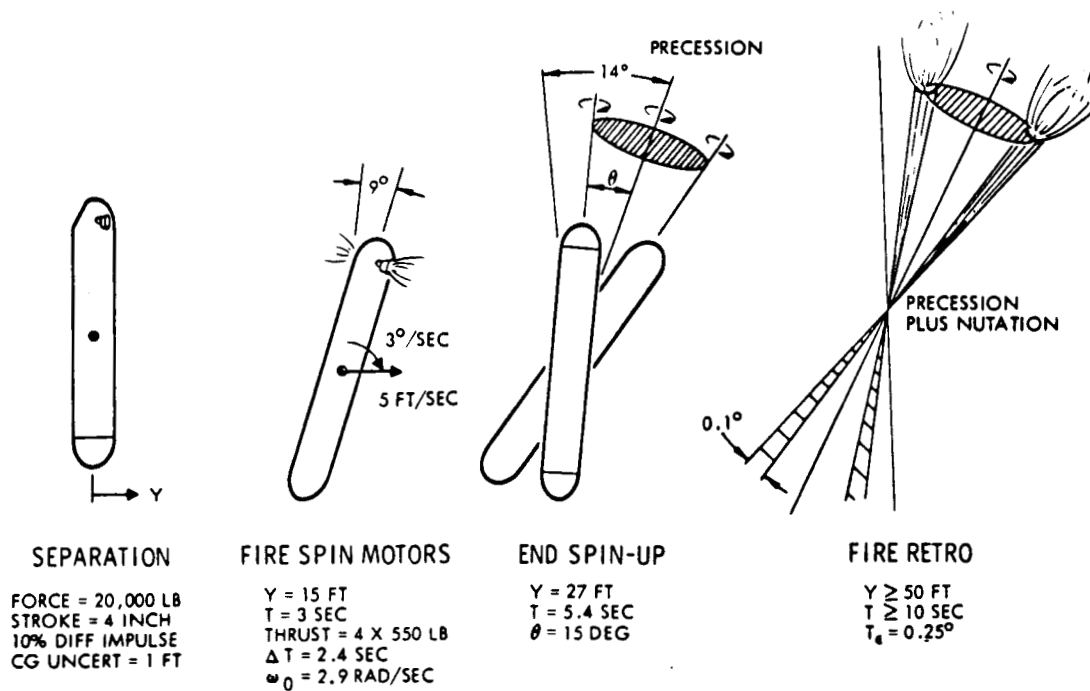FIRE RETRO

Y ≥ 50 FT
T ≥ 10 SEC
T_d = 0.25°

Figure 4-26. Tank Separation Dynamics, Tank Disposal

and pitch or yaw attitude rates are terminated. Ignition of the retro motors may therefore be delayed without influencing the tank attitude error and thus ensure tank clearance from the orbiter.

4.3.3.2 Separation System Design

The installation of the tank on the orbiter and the provisions for tank attachment, separation, and jettison are shown on Figure 4-27. As noted, the structural attachments are combined with the separation mechanism and are at stations 1087 and 1887 on the orbiter. The aft tank drag strut is reacted by the orbiter main thrust structure. Pyro actuators are installed in the orbiter at the lower diagonal struts; and when activated, they stroke the strut tank assembly for four inches to impart velocity for tank separation. Redundant circuits to redundant igniters are utilized for redundancy. A mechanical and an explosive separation device is presented in Figure 4-28. The decision for the use of either design is deferred to the Phase C portion of the program.

4.3.4 TANK ENTRY DYNAMICS AND TRAJECTORIES

One of the key questions about the OEHT concept concerns the acceptability of tank disposal. The constraints imposed by land mass impact or probability of tank/ship impact were evaluated. Secondly, the nominal tank trajectory and impact dispersion were evaluated.

The selected operating procedure for tank disposal is:

1. Inject orbiter (with tanks) into 50 x 100 nautical-mile orbit

2. Separate tanks during orbit coast

3. Deorbit tanks

4. Initiate deorbit such that tanks/fragments are targeted for Indian Ocean impact.

The nominal impact point for launches from ETR or WTR with a 180-degree range angle, the nodal point for all launch azimuths, are shown in Figure 4-29. Also shown in the figure is the shipping concentration projected for 1980. The values shown represent the number of ships projected for a 5-degree x 5-degree section (≈300 x 300 nautical miles). As can be seen in the figure, the nearest land mass for launches from ETR is approximately 1400 nautical miles away. The average shipping density is shown on the right of Figure 4-29.
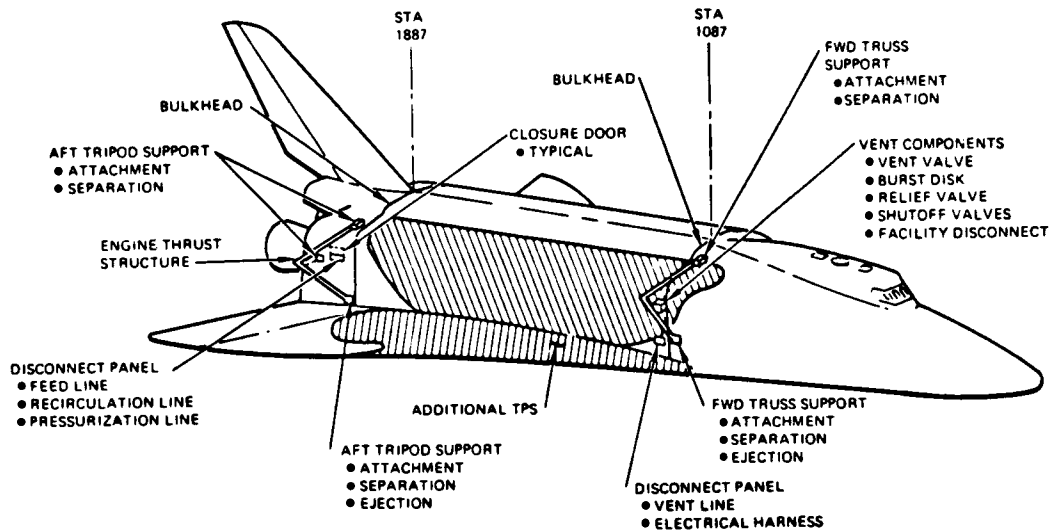
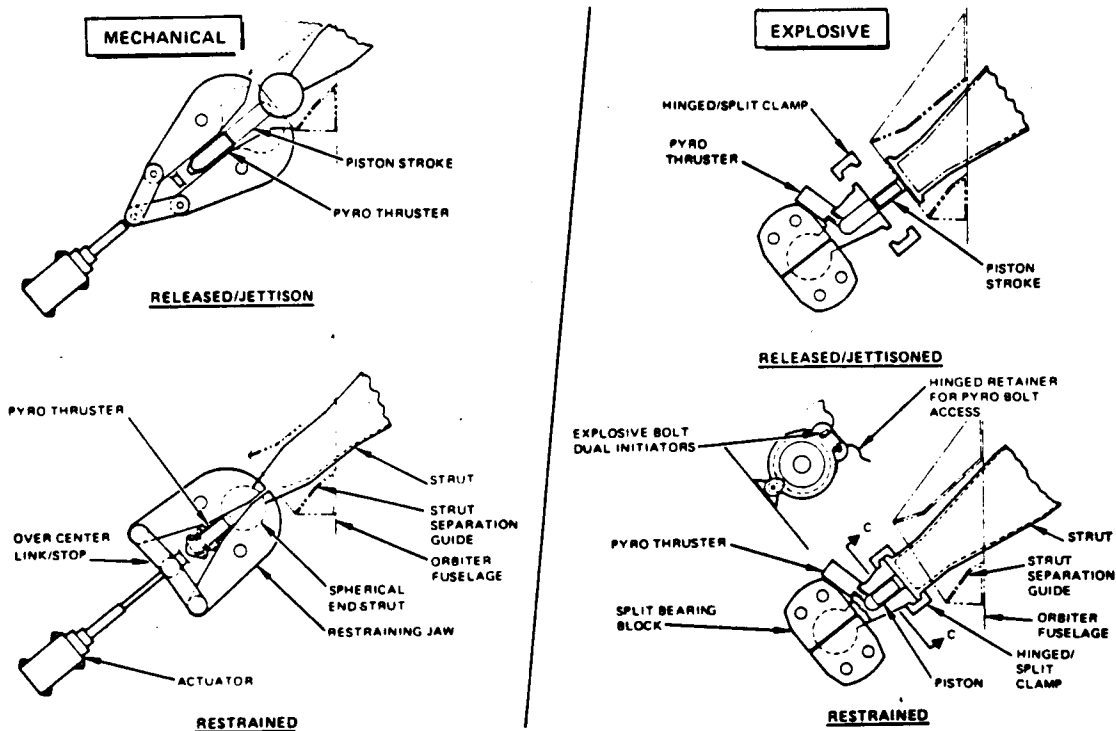Figure 4-27. Orbiter Provisions for External Tank Installation

Figure 4-28. Tank Attachment/Separation From Orbiter

The nominal tank trajectories for a normal and once-around abort trajectory are shown in Figure 4-30. As can be seen, the tanks can be targeted for the same impact range by adjusting the deorbit initiation time. The nominal mission time is 23.5 minutes after injection into the 50 x 100 nautical-mile orbit. A 300 feet per second deorbit ΔV was selected to minimize impact range dispersion errors. The trajectory parameters indicate that the tank may break up between 250,000 to 350,000 feet, unless designed to preclude breakup.

Factors which can cause impact range dispersion are (1) injection orbit errors, (2) retro timing errors, (3) separation dynamics, (4) retro thrust direction errors, (5) retro ΔV errors, and (6) aerodynamic variations. The expected tank/fragment impact range errors are shown in Figure 4-31. The range dispersion is ±600 nautical miles, which is considered acceptable in that it avoid land mass and provides a tank/ship impact probability better than one in a million.

4.3.5 TANK LOADS

The external tanks are designed primarily by internal pressure and for the external loads experienced during boost. The tanks are not designed for entry conditions, since they are allowed to fail at approximately 350,000 feet where the tank segment dispersion is acceptable.

1980 PROJECTED OVERALL WORLDWIDE MERCHANT
SHIP DISTRIBUTION (INCLUDING FISHING VESSELS)
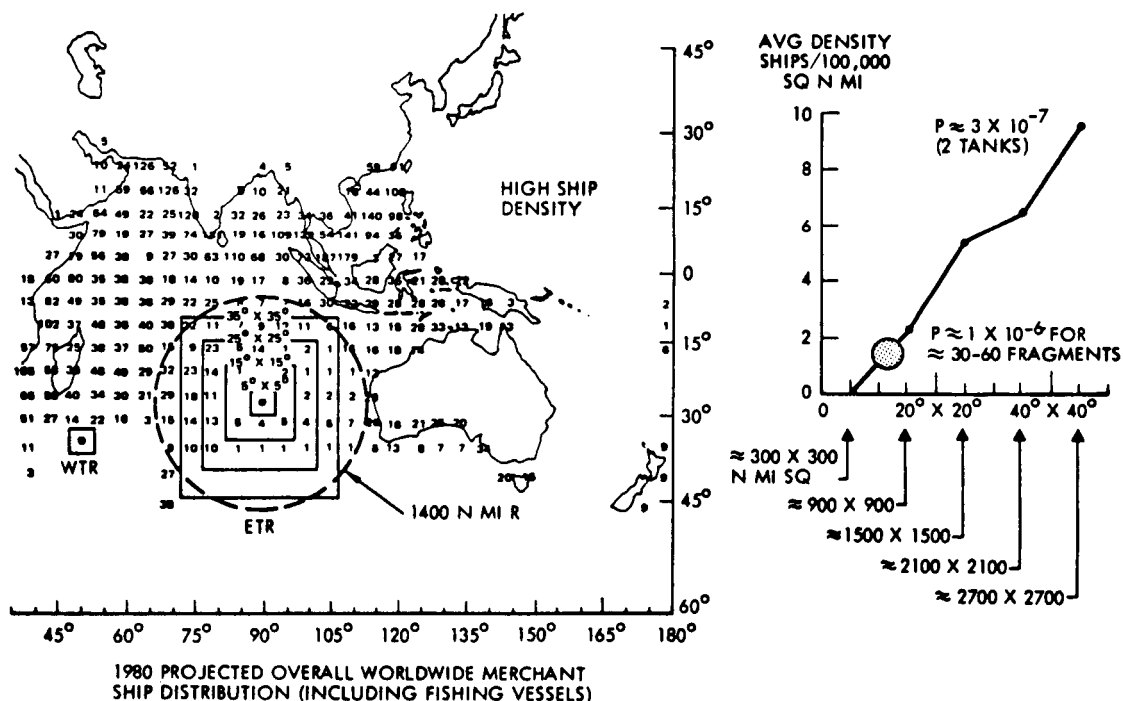
Figure 4-29. Acceptable Impact Area

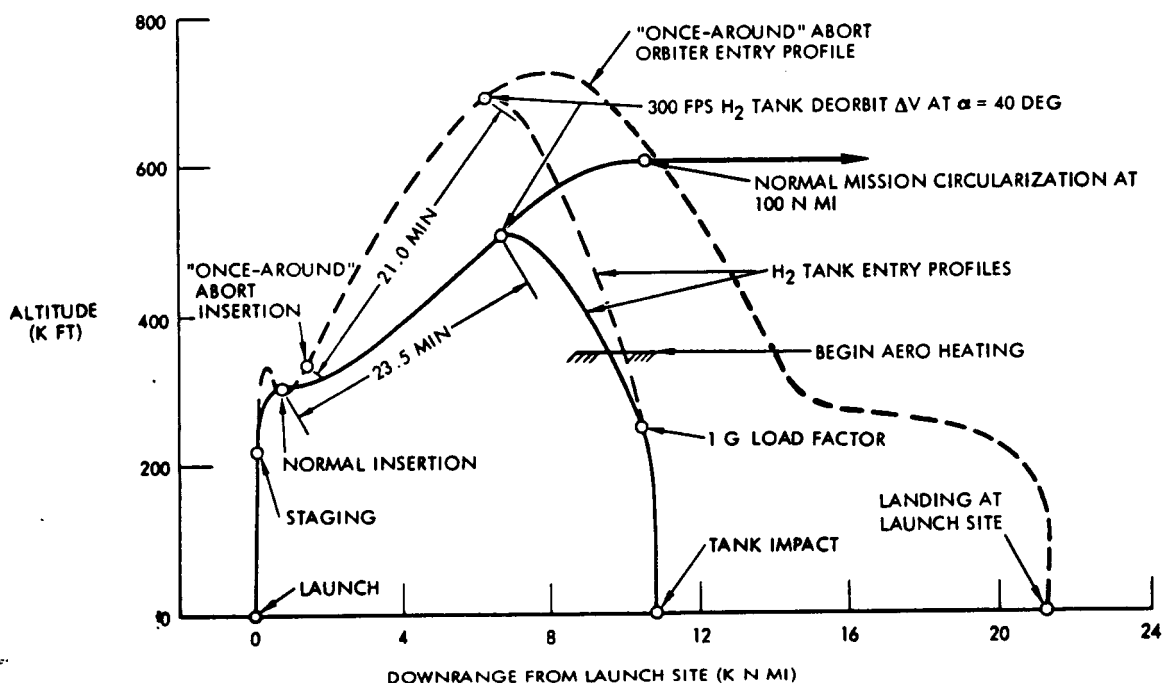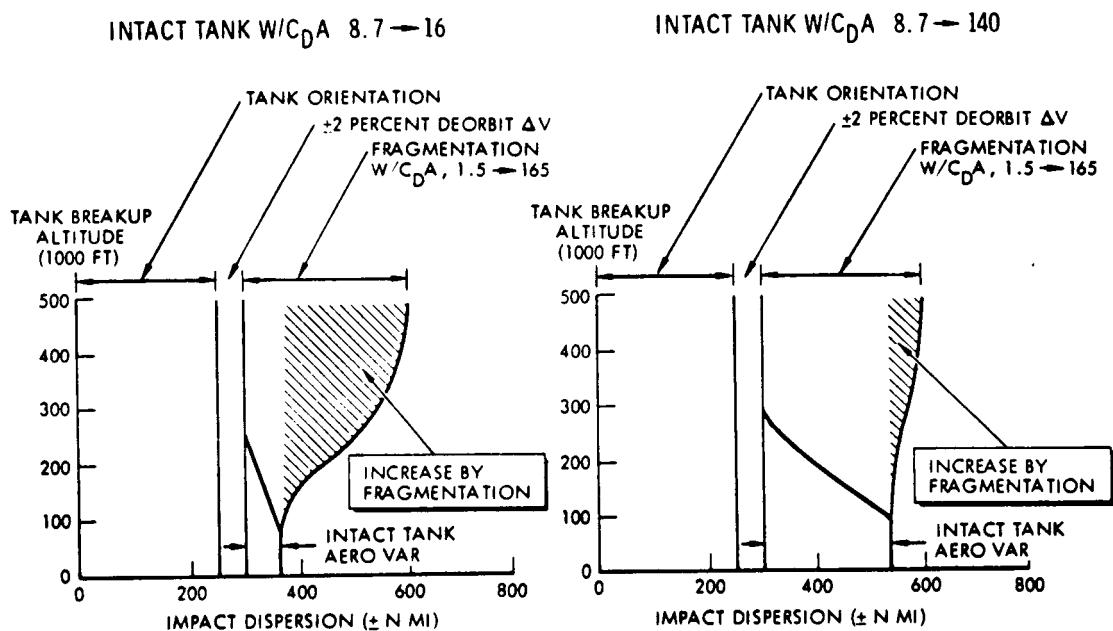Figure 4-30. Orbiter and External LH₂ Tank Flight Profiles, Normal and Abort Missions

• MAX DISPERSION ≈ ±600 N MI, ACCEPTABLE

Figure 4-31. LH₂ Tank Impact Dispersion, Fragmented Tank

4.3.6 TANK THERMAL ENVIRONMENT

The LH$_2$ external tank thermal environment was established to support the tank design definition and weight analysis. The tank thermal protection system requirements are (1) cryogenic storage requirements and (2) thermal protection of the tank through deorbit motor burn from the ascent-induced environment. Trade studies established that protecting the tank during entry to minimize tank fragment impact dispersion is not necessary.

To meet the cryogenic storage requirement of LH$_2$, three-quarter inch spray-on foam is applied to the external surface of the tank. Trade studies also considered the use of internal spray-on foam and this concept was found to be inferior.

Thermal analysis was conducted to establish any additional TPS requirement. The criteria considered was (1) the maximum allowable spray-on foam surface temperature is 500 F; (2) if an ablator is applied over the foam, the maximum bondline temperature allowable is 300 F; and (3) where ablator is applied on nontank structure (such as nose fairing), the aluminum maximum allowable temperature is 300 F through boost and 500 F through deorbit motor burn. The TPS requirements (unit weight) are shown in Figure 4-32. Ablator TPS is required on the nose fairing. Ablator is also required on the side of the tank adjacent to the orbiter because of the ascent interference heating environment.

4.3.7 TANK STRUCTURE AND TPS

The tank structure, including the pressurized tank, the forward fairing, and the forward and aft support structure were designed to the loads schematically depicted on Figure 4-33. As noted, the propellant inertia, the aerodynamic lift, and the internal pressure are the applied loads. The resulting loads applied to the tools are 35 psi average pressure and 372,000 lbs longitudinal reaction. The basic design of the tank is monocoque, with the support structure strut reactions at the ends only. The forward fairing is of aluminum skin stringer and frame construction.

The design conditions which determined the minor elements of the structure and the various areas of the tank are also noted on the figure. The skin thickness in the spherical portions of the end bulkheads has been increased to .080 inch to allow unpressurized ground handling and handling during prelaunch when unfueled. The tank must be pressurized for flight, transportation, prelaunch (when fueled), launch, and entry.

The tank structure is shown on Figure 4-34 and, as noted, is a monocoque shell structure welded together from a preassembled cylinder and two end bulkheads. The tank skins are chem-milled to provide the

- UNIT WTS (PSF) SHOWN, EXCLUDING SEALER & BOND MATERIAL
- EXTERNAL FOAM, T_{MAX} = 500 F
- ABLATOR FOAM BOND LINE, T_{MAX} = 300 F
- ABLATOR/AL BOND LINE, T_{MAX} = 500 F

DETAIL A

TOTAL TANK TPS WEIGHT = 4120 LB

SECTION A-A   **SECTION B-B**   **SECTION C-C**
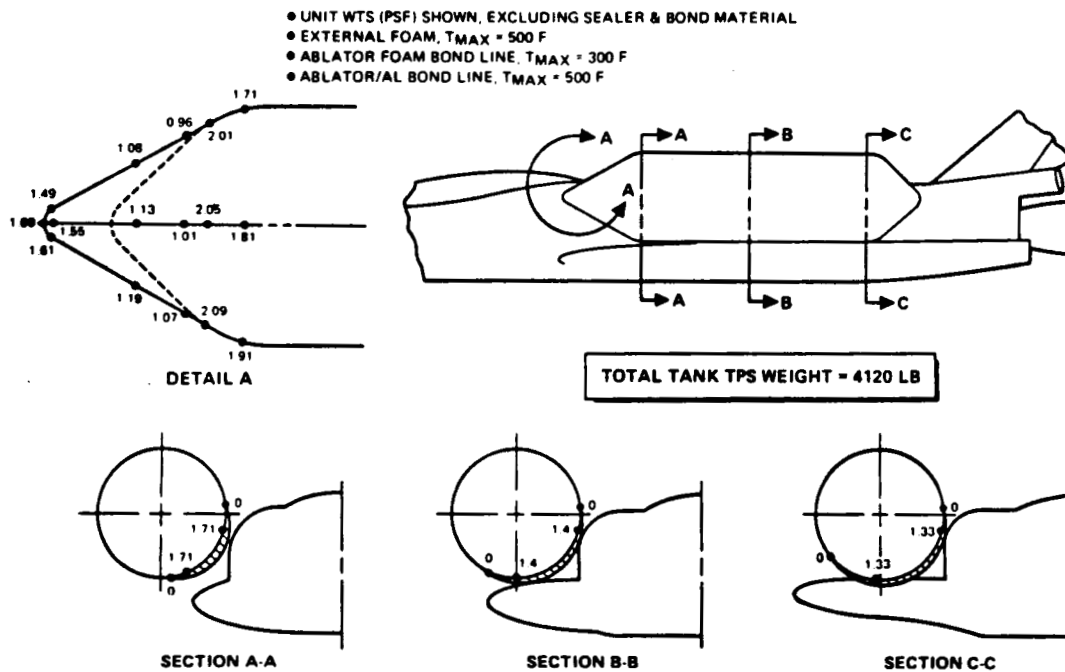
Figure 4-32. Baseline Tank TPS Requirements

- LOADS/LIMIT

BASIC DESIGN
- TANK MONOCOQUE ALUMINUM
- STRUTS TANK END REACTIONS
          TITANIUM TUBES
- FAIRING SKIN STRINGER FRAME (ALUM)

DESIGN CONDITIONS
A   AERODYNAMIC PRESSURE (COMPRESSION)
B   STRUT REACTION R_1 (SHEAR)
C   HOOP COMPRESSION (UNPRESSURIZED)
D   HOOP TENSION (PRESSURIZED)
E   HOOP COMPRESSION (UNPRESSURIZED)
F   STRUT REACTION R_2 & R_3 (SHEAR/COMPRESSION)
R_1 $\begin{cases} \text{TENSION (LOWER STRUT)} \\ \text{COMPRESSION (UPPER STRUT)} \end{cases}$
R_2 TENSION (BOTH STRUTS)
R_3 COMPRESSION

GROUND CONDITIONS UNPRESSURIZED
- GROUND HANDLING   } HORIZONTAL
- PRE LAUNCH (UNFUELED) } VERTICAL

FLIGHT OR PRESSURIZED CONDITIONS
- TRANSPORTATION
- PRE LAUNCH (FUELED)
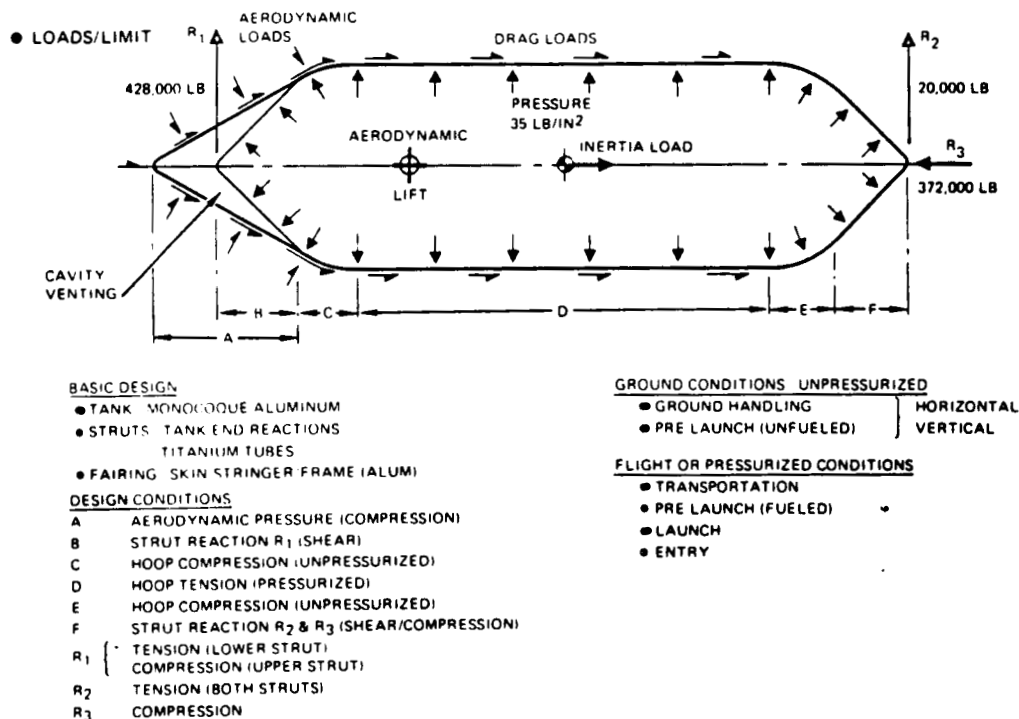- LAUNCH
- ENTRY

Figure 4-33. LH$_2$ Tank Structural Design Loads (Limit)

proper skin thickness as reduced from the 0.160-inch weld lands. The end bulkheads contain similar sized manhole covers which are utilized for tank close-out, propulsion lines installation, and the attachment of the end support structure. The support structures are an aft-fixed tripod of titanium tubes and a forward titanium-tube hinged truss which provides a determinate structure and allows tank deflection growth, etc., relative to the orbiter. The forward fairing is attached directly to a ring with tapped bosses welded into the forward bulkhead. The forward fairing has a removable forward section to provide access to the equipment (roll motors, deorbit motor, etc.) installed in the fairing.

The pressurized LH_2 tank was designed considering the fracture mechanics of the cryogenic fuel tank. The externally mounted reusable LH_2 tank for the OEHT 0500 orbiter has different service life requirements compared to the baseline 161C orbiter. Figure 4-35 defines these service life differences and the significant factors resulting from the differences. As noted, to detect the larger allowed skin flaw sizes (cracks) the external LH_2 tank can be proof pressure tested at a lower pressure. The leakage and proof test demonstration and inspection should be easier; and therefore, the cost of the tanks with respect to fracture mechanics should be slightly less than for the external tank. A basic program evaluation is that the risk factor for the successful operation of the LH_2 tanks should be lower for the external tank orbiter than for the baseline 161C fully reusable orbiter.

The design for the LH_2 tank TPS is shown on Figure 4-36. The SOFI is installed on the exterior surface of the tank and is partly covered by cork ablator over the forward end and the inboard quadrant where the tank is adjacent to the orbiter body and wing. The SOFI utilized requires a new development to retain its physical integrity at 300 F. The present SOFI, as utilized on the S-II tank, is usable only to 200 F. The 300 F upgrade in temperature limitation is compatible with the SOFI being proposed for the baseline 161C orbiter integral (internal) LH_2 tank. The individual struts of the forward and aft support structures and the large diameter feed line are also covered over their forward areas by a high density ablator which will provide thermal protection during boost.

A SOFI installation trade study was developed at the start of the preliminary design phase to determine whether the SOFI would be installed inside or outside the tank skins. The revision to the TPS requirement for tank protection during boost only (with inherent capability to enter to 350,000 feet) reduced the amount of cork ablator required, and therefore, a possible design revision was evaluated which would reduce program costs. Figure 4-37 illustrates the study and provides the results. The tank assembly with an interior SOFI installation is heavier than the tank with an exterior SOFI installation—and the basic reason for the difference is the temperature limitations of the SOFI which required a decreased tank skin structural
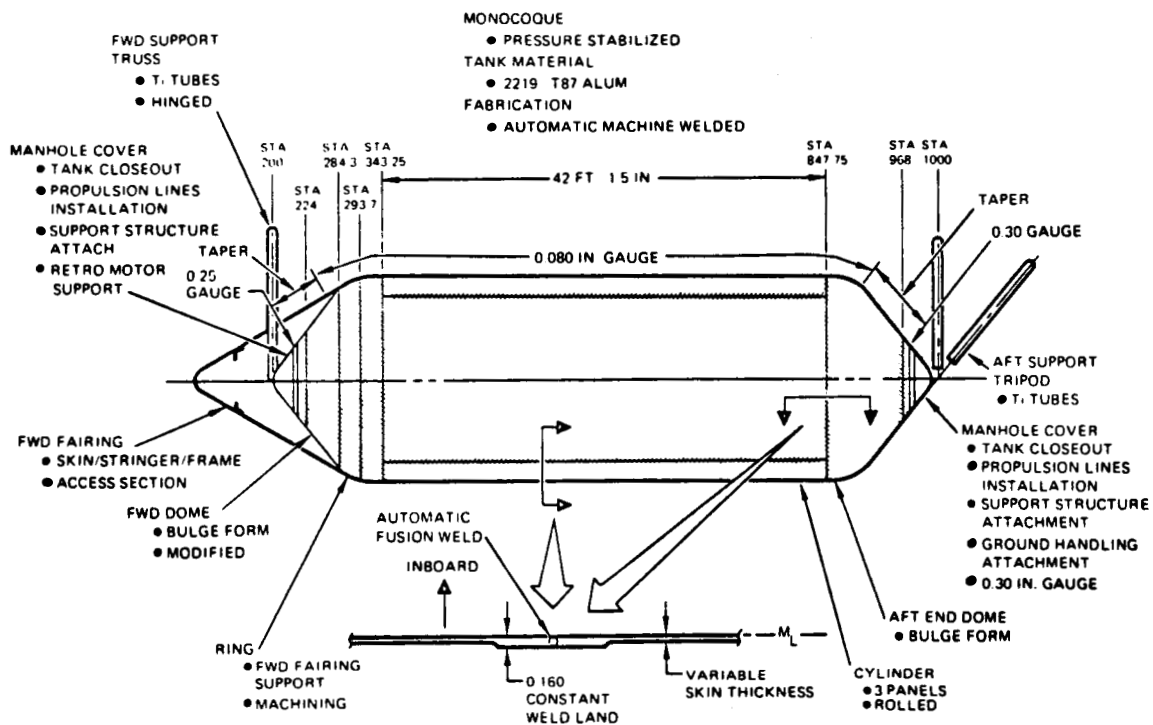
Figure 4-34. LH₂ Tank Structure

Figure 4-35. Fracture Mechanics Consideration Differences

Figure 4-36. LH₂ Tank TPS

Figure 4-37. External LH₂ Tank SOFI Installation Study

allowable and a greater TPS cork coverage. Basic assumptions were (1) no failure of internal SOFI, prior to fuel dump and, (2) no failure of outside SOFI prior to tank jettison. This was required to provide protection for the engine hydrogen intake against internal SOFI falling into the propellant dumped through the engine. The exterior SOFI installation was selected because of lesser tank assembly weight and lower program costs.

4.3.8 TANK PROPULSION AND FLUID SYSTEMS

The external tank propulsion and fluid system provides LH_2 feed to the orbiter main propulsion system during main engine operation and provides spin stabilization and deorbit impulse after tank separation.

The interface of the external LH_2 fluid tanks with the remainder of the orbiter main propulsion system is essentially the same as for the internal LH_2 tank of the 161C configuration except that the interface occurs at a disconnect panel for tank/orbiter separation. Four fluid lines are provided for (1) tank fill, drain, and engine feed, (2) tank pressurization and boil-off control, (3) LH_2 recirculation for engine subcooling, and (4) tank venting during fill and prelaunch operations. All four lines interface with corresponding lines on the orbiter.

Tank separation and disposal requires that the tanks be spin stabilized and a deorbit velocity provided. This capability is provided by four solid rocket motors for tank spin up and a single solid rocket motor for deorbit impulse.

A trade study was performed to determine whether the roll motors could be eliminated using a shorter firing time retro motor and still provide acceptable tank orientation and clearance with the orbiter during retro. The options and results of the study are shown on Figure 4-38 and as indicated, the use of the roll motors and the deorbit motor requires two systems but they are lighter in weight. The use of the shorter firing time deorbit motor utilizes only one system--but is heavier in weight. Elimination of the roll motors increases tank dispersion sensitivity to tank cg range and variations in impulse from the two separation actuators. (The impact of the two tank separation and deorbit option on dispersion is documented in Section 4.3.4 of this report.) The baseline separation and deorbit system design for the tank utilizes a spin system. It is however considered that follow-on studies should continue to examine the two previously described options.

DEORBIT MOTOR + ROLL MOTORS SYSTEMS

DEORBIT MOTOR
- IMPULSE   114,000 LB SEC
- FIRING TIME ~28 SEC

ROLL MOTOR
- IMPULSE · 1320 LB SEC
- FIRING TIME ~2 SEC

ORIENTATION TOLERANCE
- ±30°

DEORBIT MOTOR SYSTEM

DEORBIT MOTOR
- IMPULSE   114,000 LB SEC
- FIRING TIME ~5 SEC
- INCREASED DEVELOPMENT COSTS

WEIGHT INCREASE
- ~100 LB (PER TANK)

ORIENTATION TOLERANCE
- ±30°

PROBLEM
- EXTRA SYSTEM VS EXTRA WT
- ADDED DEVELOPMENT COSTS
- BASELINE ROLL SYSTEM & REEVALUATE IN STUDY EXTENSION

FWD SUPPORT STRUTS
FWD FAIRING
ROLL MOTORS
DEORBIT MOTOR
PROPELLANT TANK
37
30°
STA 140
STA 200
PROPELLANT TANK
DEORBIT MOTOR
53
30°

Figure 4-38.   External Tank Attitude Control Trade Study

4.3.9 MASS PROPERTIES

The summary weight statement for the external LH_2 tanks is given in Table 4-7. The weight statement is for two tanks, the forward 60-degree cone-fairing without side fairing, and the exterior installed SOFI.

4.3.10 TANK EVALUATION

A summary of the tank concepts that were developed during the Phase 2 portion of the study and incorporated into the preliminary design of the exterior LH_2 tanks is given on Table 4-8. The selected concepts of the various trade studies were selected on the basis of minimum weight and minimum program costs.

The preliminary design of the external LH_2 tanks was done with engineering, manufacturing, and program requirements in mind. Ease of manufacturing, accessibility to equipment, and reduced costs were considered. The retainment of equipment in the orbiter rather than in the ejected tank allowed the reuse of the equipment. Increasing the gauge of part of the tank skins allowed the handling of the unpressurized tank assembly

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Table 4-7. External Tank Launch Summary Weight Statement

CODE	SYSTEM	A	B	C	D	E	F
CONFIGURATION: EXTERNAL HYDROGEN TANK			BY NR		DATE		
1.0	WING GROUP						
2.0	TAIL GROUP						
3.0	BODY GROUP	11265					
4.0	INDUCED ENVIR PROTECT	5610					
5.0	LANDING, DOCKING						
6.0	PROPULSION, ASCENT	1610					
7.0	PROPULSION, CRUISE						
8.0	PROPULSION, AUXILIARY	580					
9.0	PRIME POWER						
10.0	ELECTRICAL CONV & DIST						
11.0	HYDRAULIC CONV & DIST						
12.0	SURFACE CONTROLS						
13.0	AVIONICS						
14.0	ENVIRONMENTAL CONTROL						
15.0	PERSONNEL PROVISION						
16.0	RANGE SAFETY						
17.0	BALLAST						
18.0	GROWTH	1850					
19.0							
	SUBTOTAL (DRY WT)	20915					
20.0	PERSONNEL						
21.0	CARGO						
22.0	ORDNANCE						
23.0	RESIDUAL FLUIDS	298					
24.0							
	SUBTOTAL (INERT WT)	21213					
25.0	RESERVE FLUIDS	928					
26.0	INFLIGHT LOSSES	737					
27.0	PROPELLANT-ASCENT	112893					
28.0	PROPELLANT-CRUISE						
29.0	PROPELLANT-MANEUV/ACS						
30.0							
	TOTAL WEIGHT - LB	135771					

DESIGNATIONS:

ITEM
A
B
C
D
E
F

NOTES & SKETCHES

Table 4-8. External LH$_2$ Tank Phase 2 Selection Summary

Concept	Relative Weight	Complexity	Comment	Selection
Tank Forward Fairing				
60 degree cone-fairing	Light	Greater	Weight advantage is small	✓
60 degree cone-integral tank	Heavier	Less	Design advantage could override	
45 degree cone-integral tank	Lightest	Least	High boost drag	
Tank to Orbiter Fairing				
No side fairing	Heavy	Less	Reusable TPS	✓
Side fairing	Heavy	Greater	Large throw-away cost	
SOFI Temperature				
200 F	Heavier	Same	Presently available	✓
300 F	Lighter	Same	Development req.	
SOFI Installation				
Internal	Heavier	Greater	Temperature limitations of SOFI	✓
External	Lighter	Less	Temperature limitation of SOFI	
Tank Separation/Ejection				
In struts	Same	Greater	Greater throw-away costs	✓
In orbiter	Same	Less	Reusable mechanism system	

Table 4-8. External LH$_2$ Tank Phase 2 Selection Summary (Cont)

Concept	Relative Weight	Complexity	Comment	Selection
Tank Disposal				
Entry protected	Heavy	Greater	Breakup at ~ 200,000 ft ⎤ Acceptable	
No entry	Lighter	Less	Breakup at ~ 350,000 ft ⎦ dispersion	✓
Vent System				
Tank	Heavier	Greater	Larger throw-away costs	
Orbiter	Lighter	Less	Reusable components	✓
Propellant utilization				
PU	Heavier	Greater	Greater throw-away costs	
No PU	Lighter	Less	Added propellant bias	✓
Tank Orientation				
Roll motors	Lighter	Greater	Two systems	Resolution Deferred
No roll motors	Heavier	Less	Greater dispersion sensitivity	

during manufacturing and of the orbiter (unfueled) in either the horizontal or vertical position. A listing of the design features for the reduction in program costs is as follows:

1. SOFI on exterior of tank rather than interior of tank

2. Monocoque tank structure pressure stabilized rather than stiffened

3. Simple chem mill pattern for weld lands

4. Cylinders of three maximum size aluminum sheets rather than many conventional size sheets welded together

5. Elimination of PU system; use of sensor installation for fill

6. Tank end reaction for load inputs

7. Constant weld land thickness

8. Common access manhole diameter

9. Common front and aft bulkhead shapes—use of same forming tool

10. Increased skin gauge (040 to 080) in spherical area of the bulkheads (handling either vertical or horizontal in unpressurized state)

11. Release, attach, and ejection mechanisms in orbiter (reused)

12. Front end of forward fairing removable for access to installed equipment

13. Vent valve system installed in orbiter (reuse)

14. Self-sealing (not motor driven) disconnect on tank side

4.4 SELECTED BOOSTER DEFINITION

This section summarizes the characteristics of the booster which were analyzed in Part II of the study (B17E). The starting point for this booster definition was the selected booster from Part I of the study. Results of these analyses were used in the final sizing, discussed in Section 4.5.

4.4.1 BOOSTER DESIGN

The selected booster configuration is presented in Figure 4-39. Except for being slightly smaller, it is similar to the heat sink booster of Part I of the study, described in Section 3.5. It is basically derived from the Phase B baseline, B-9U, with the necessary changes to make it a heat sink booster. It has a low delta wing, single vertical tail, and forward-mounted canard. The body is basically a cylinder with the tanks and intertank section having sufficient wall thickness to serve as heat sinks, thus eliminating much of the reradiative heat shields. Fairings are required to streamline the canard/body and wing/body intersections. The orbiter is attached to the upper surface of the booster with a link system
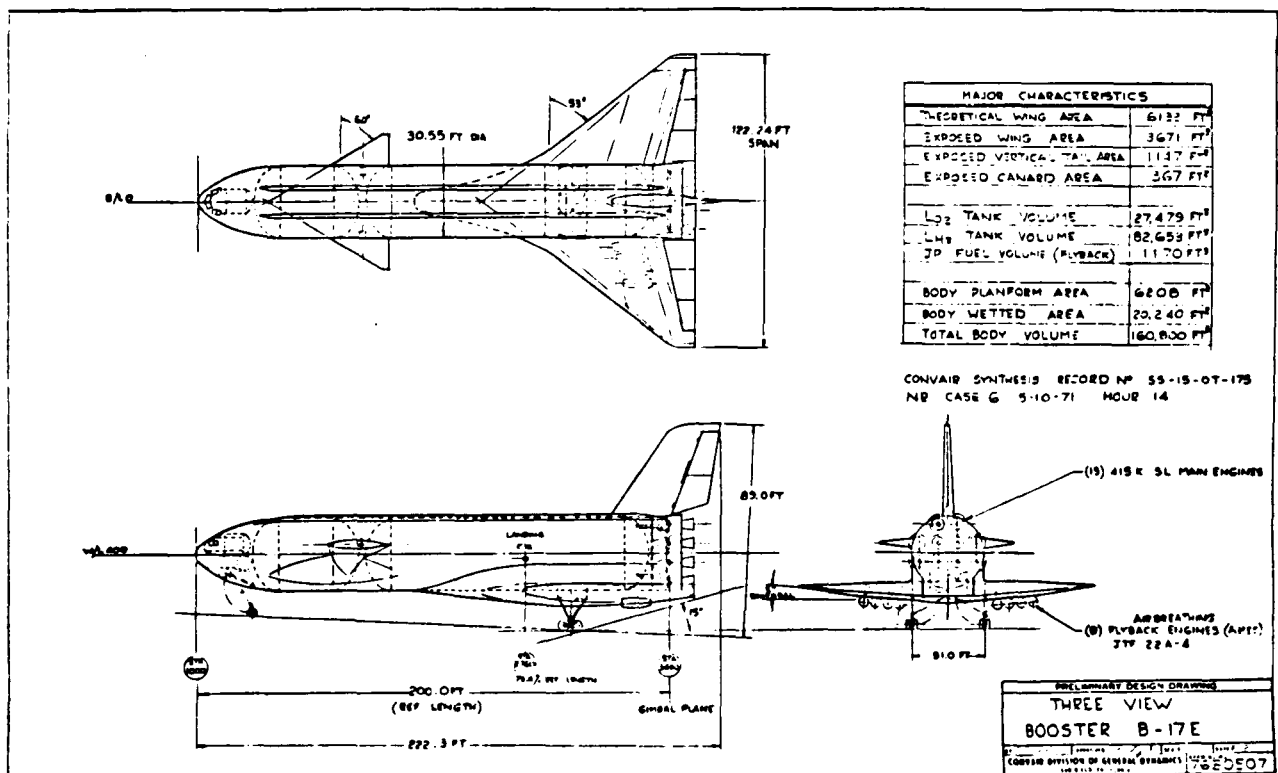

Figure 4-39. Three-View of Booster B-17E

that also serves as the separation system. In an attempt to obtain tail clearance with the larger orbiter and smaller booster, it was necessary to move the forward orbiter attach support into the LO_2 tank. The LO_2 lines and other subsystem lines are routed on the upper surface of the vehicle within fairings.

The main propulsion system consists of thirteen 415,000-pound sea level static-thrust LO_2/LH_2 engines installed in the base of the vehicle. Eight JTF-22A turbofan engines are deployed below the wing for booster flyback propulsion.

Internally, the vehicle is arranged with the crew compartment and avionics bay located in the nose, a forward LO_2 tank, and an aft LH_2 tank. The tanks provide the primary load-carrying structure of the booster. The tanks are joined by a cylindrical intertank structure that supports the canards and the forward orbiter attach drag link. The tanks and intertank are of internal frame stringer construction, and the outside wall provides a smooth aerodynamic surface. The wall thickness is increased above the strength requirements to provide sufficient material for heat sink.

The delta wing has the same planform and orientation as the Phase B baseline. The wing is sized to yield a landing wing loading of 75.6 lb/ft^2. The wing lower surface is a titanium heat sink, and the upper surface is a corrugated titanium heat shield. The wing-thickness ratio at the outboard air-breathing engine is increased over that of the Phase B baseline to accommodate the fixed-sized air-breathing engines.

The vertical tail and canard surfaces have the same geometry as the Phase B baseline booster. Both of these surfaces are heat sink.

The flyback fuel is contained in two tanks: one large tank in the wing carry-through and a smaller tank forward of the LO_2 tank. This arrangement, which is consistent with the Phase B baseline, was chosen to yield a forward c.g. for entry.

The various vehicle subsystems are located in locations similar to those of the B-9U. Any subsystem lines that must traverse the body are rooted on the upper surface within fairings.

4.4.2 BOOSTER AERODYNAMICS

The aerodynamic characteristics of the B-17E booster configuration are summarized in Figures 4-40 and 4-41. These characteristics are based on experimental wind tunnel data obtained in the Phase B test program for the B-9U baseline configuration. The geometric differences between the wind tunnel models and the B-17E configuration have been adjusted through
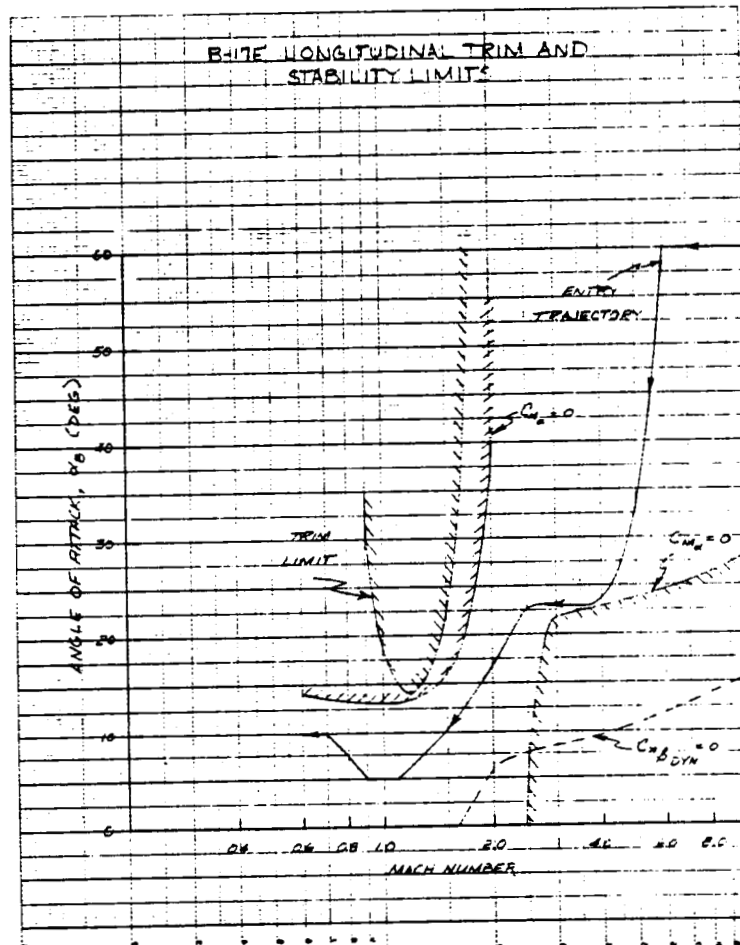
- 87 -

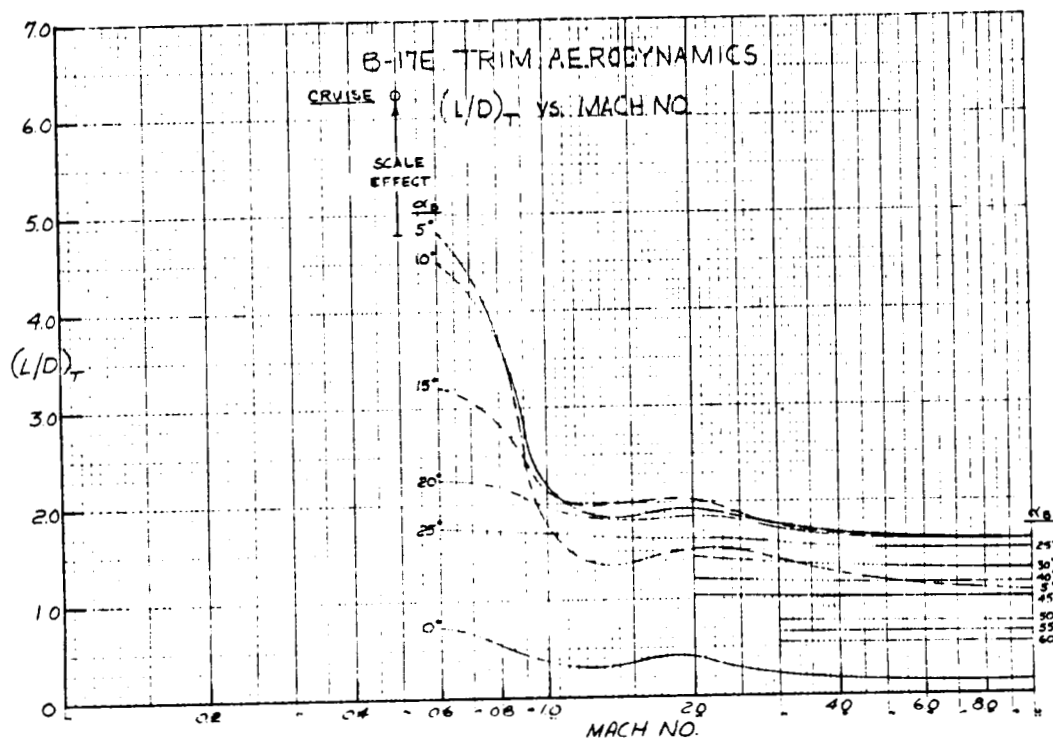Figure 4-40. B-17E Longitudinal Trim and Stability Limits

Figure 4-41. B-17E Trim Aerodynamics

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the use of component buildup data. However, because these differences were minor, the aerodynamic characteristics of B-17E are nearly identical to those of the B-9U.

The trim and stability boundaries for the B-17E are presented in Figure 4-40 as a function of angle of attack and Mach number. Also shown in this figure is a trace of the nominal B-17E entry trajectory, which is seen to be constrained primarily by the longitudinal stability boundaries ($C_{m_\alpha} = 0$). Although the B-17E configuration exhibits static directional instability above Mach = 1.1, it is found to have positive dynamic C_{n_β} throughout the trajectory.

Figure 4-41 presents the trimmed lift-to-drag (L/D) ratio characteristics of the B-17E as a function of Mach number. The average flyback cruise L/D indicated in this figure is essentially the same as for the B-9U.

4.4.3 BOOSTER PERFORMANCE

The B-17E nominal mission profile is shown in Figure 4-42. In concept, this profile is similar to that of the B-9U. However, the use of a three-engine orbiter with external LH$_2$ tanks significantly reduces the staging velocity, resulting in a shorter and less severe entry and a shorter flyback distance. The flyback distance of 263 n. mi. requires 55,907 pounds of fuel (one engine out, directional headwinds), compared with 399 n. mi. and 116,357 pounds of fuel for the B-9U.

The landing characteristics of the B-17E are quite similar to the B-9U because the aerodynamic characteristics are similar and the wing has been sized to give the same landing wing loading of 75.6 lb/ft^2, based on theoretical wing area and empty landing weight.

The stage separation system for the B-17E is the same link concept used on the B-9U, with only those changes dictated by the change in vehicle size. Separation characteristics at nominal staging were analyzed for various numbers of orbiter engines operating. The results, shown in Figure 4-43, indicate satisfactory separation in all cases.

The ferry performance for the B-17E and B-9U is presented in Table 4-9. From this table it can be seen that the ferry range capability of the B-17E is much less than that of the B-9U, because of its lower fuel capacity. However, the only mission not met by the B-17E is the hot-day condition from Kennedy Space Center to Robbins Air Force Base, which requires a 300 n. mi. ferry capability.

The abort procedures for the B-17E are similar to those of the B-9U. A brief analysis of max qα separation indicated that safe separation could
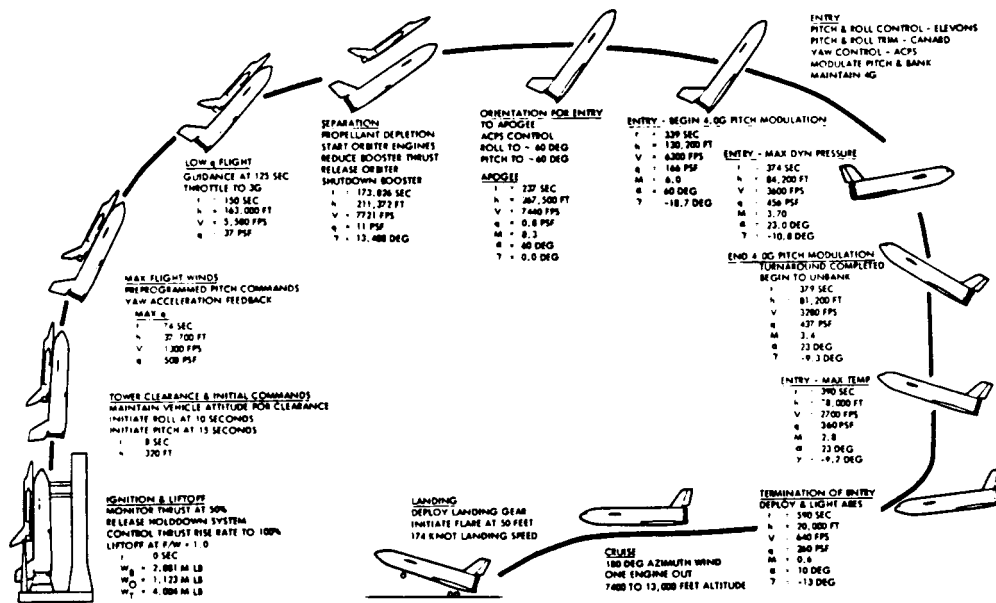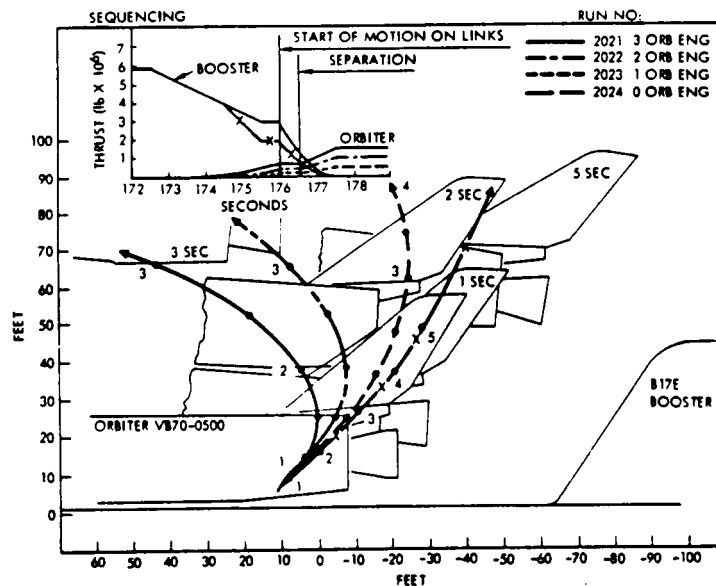
Figure 4-42. Booster Nominal Mission Profile

Figure 4-43. Separation Characteristics

not be obtained using only orbiter TVC with a gimbal angle of ±5 degrees. Increased orbiter gimbal angle or the use of aerodynamic surfaces would probably provide satisfactory abort separation. As with the Phase B baseline, there would be a weight penalty associated with structural beefup of the orbiter, booster, and links to achieve max qα separation.

4.4.4 BOOSTER CONTROL

Booster flight control is required for all phases of flight from liftoff through landing. Flight control is obtained primarily with main engine gimballing during ascent, ACPS engines during initial entry, and aerodynamic surfaces during final entry and subsonic flight. Since the aerodynamic characteristics and configuration of the B-17E alone are similar to those of the B-9U, it was assumed that the general aerodynamic flight control characteristics developed for the Phase B baseline B-9U were applicable to B-17E.

Results of detailed ascent control simulations indicated that the control requirements for the B-17E during the region of high aerodynamic activity were more severe than for the B-9U. These results were principally due to the change in lateral aerodynamic characteristics produced by the larger orbiter mated with a smaller booster.

Table 4-9.   Ferry Performance

Takeoff Conditions	Ferry Range Capability	
	B-17E (n. mi.)	B-9U (n. mi.)
Sea Level — Standard Day	339	720
Sea Level — Hot Day	155	390
4000 Feet — Standard Day	339	720
4000 Feet — Hot Day	322	660
Sea-Level Runway: 10,000 feet 4000-Foot-Altitude Runway: 13,600 feet		

The pitch-plane requirements are similar to those indicated for the B-9U vehicle. Some trajectory wind biasing or load relief would be required to hold the gimbal angle within ±10 degrees and the max $q\alpha$ less than ±2800 deg-psf.

In the yaw plane, the problem was more severe due to the increased yaw aerodynamic stability and yaw-roll coupling. The use of aerodynamic surfaces (elevons) for roll control and some load relief would be required to hold the gimbal angle within ±10 degrees and the max $q\beta$ less than ±2400 deg-psf.

An analysis of B-17E entry performance indicated that the ACPS control requirements could be met with 26 thrusters, four fewer than required by the B-9U. The total impulse requirements were reduced only slightly.

4.4.5 BOOSTER LOADS

The booster-alone loads of the B-17E are less than those of the Phase B baseline B-9U because the vehicle is smaller. The fact that the booster is smaller, while the orbiter is larger, increases the mated-configuration loads. Table 4-10 presents a summary of the attachment loads for the B-17E.

4.4.6 BOOSTER THERMAL ENVIRONMENT

The B-17E booster is significantly different from the B-9U in the area of thermal environment and resultant vehicle effects. This is due to the fact that its staging velocity is significantly lower and it is primarily a heat-sink booster.

In a heat-sink system, the initial temperature has significant effects. For the B-17E, the LO_2 tank is uninsulated-yielding a -290 F initial temperature-and the LH_2 tank incorporates a dual-insulation concept consisting of PPO foam over the wall and Lexan plastic caps over the stiffening webs to prevent cryopumping on a cold, still day. A system having an effective PPO thickness of 0.3 inch and 0.15-inch Lexan caps was selected for the B-17E. This system yielded an initial wall temperature of -50 F on a hot, windy day and did not cryopump on a cold, still day. The aluminum body was assumed to have a maximum temperature limit of 300 F.

The heat-sink thickness requirements for the LO_2 tank intertank and LH_2 tank are presented in Figure 4-44. Canard/body and wing/body interference estimates indicated significant effects in limited regions over the body. These interference effects were accounted for in sizing the heat-sink material and the canard and wing fairings.

Table 4-10. Mated B-17E/VB70-0500 Boost Phase Separation System Logic

BOOSTER B-17E

BOOST PHASE

SEPARATION

SYSTEM LOADS

Condition		Wind	FX (x10³ lb)	FY (x10³ lb)	FZ (x10³ lb)	AY (x10³ lb)	AZ (x10³ lb)	MX (x10³ lb)
One Hour Ground Winds Fueled, Unsupported		Head	1122	0	99.6	0	108	0
		Tail	1122	0	36.4	0	178	0
		Side	1122	$\pm$27.7	80.3	$\pm$11.5	135	$\mp$5.10
Dynamic Liftoff + One Hour Ground Winds		Head	1746	0	109	0	181	0
		Tail	1744	0	102	0	254	0
		Side	1745	$\pm$1.64	118	$\pm$13.1	209	$\mp$2.68
Max. α-q	+2800	Head	2342	0	-190	0	-590	0
	-2800	Tail	2311	0	283	0	873	0
Max β-q	2400	Side	2356	$\pm$126	36.5	$\mp$279	20.3	$\pm$13.2
3G Max Thrust			3697	0	32.1	0	430	0
3G Booster Burnout			3685	0	-50.8	0	423	0

In defining the vehicle structure characteristics, the body heat-sink material was increased by 25 percent to account for heating uncertainties. The maximum temperature of the LH_2 tank lower surface centerline is presented in Figure 4-45 as a function of the heating-rate factor.

In this study, there were no design considerations for the possible formation of ice on the surface of cryogenic tanks. An analysis was made to determine the probability of that 5000 pounds of ice would form, assuming a uniform distribution and that 10 percent of the precipitation forms ice. Based on NASA's environmental criteria, and considering any day of the year, there is a 7.9-percent probability that 5000 pounds of ice would form.

4.4.7 STRUCTURE AND TPS

It is in the area of the structure and TPS that the B-17E booster is significantly different from the Phase B baseline B-9U. The relatively low staging velocity of the B-17E permits using a heat-sink booster, with a significant reduction in the amount of TPS.

Figure 4-46 presents the material distribution and peak operating temperatures for the vehicle. The less severe aerodynamic heating environment of the B-17E permits the use of less exotic materials than does the B-9U.
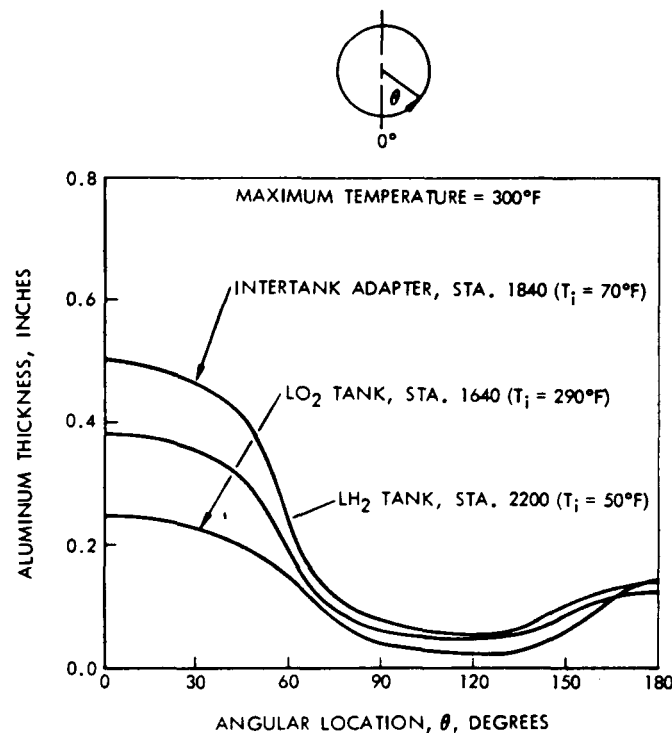

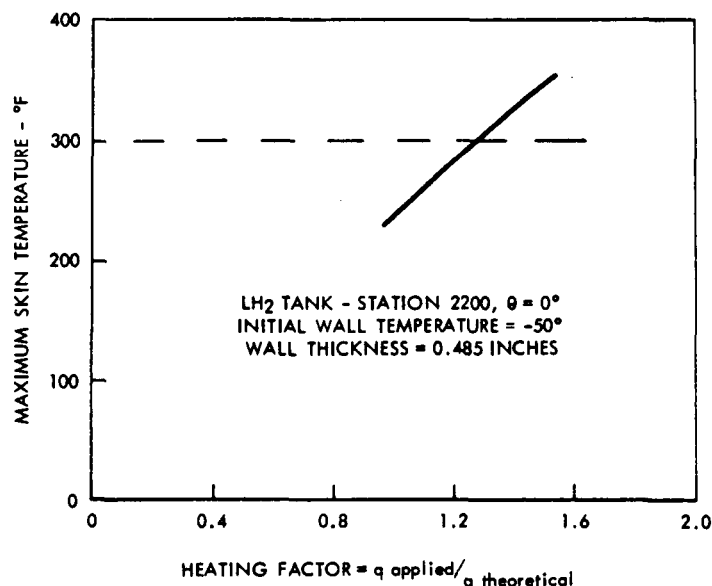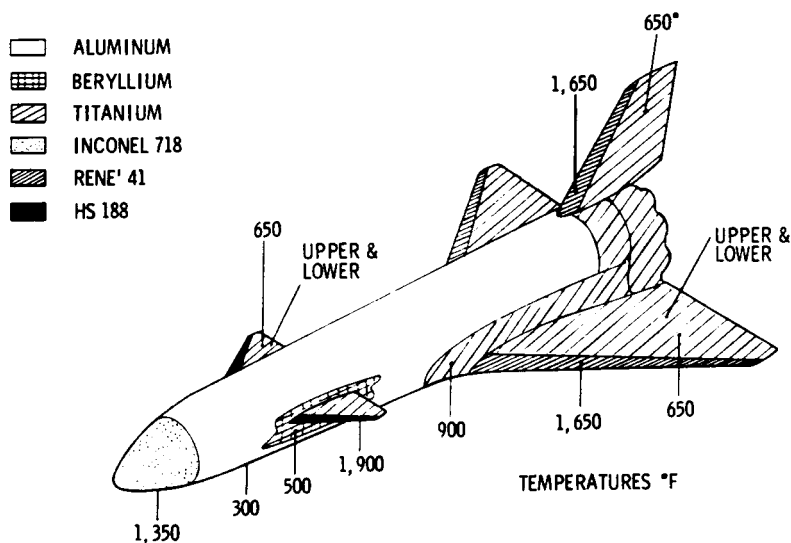Figure 4-44. B-17E Heat Sink Thickness Requirements

Figure 4-45. Heating Factor Effect on Peak Temperature

The nose structure, which contains the crew/avionics compartment, is similar in concept to that of the B-9U. It incorporates a hot structure with Inconel 718 nickel alloy skins. The forward skirt, LO_2 tank, intertank adapter, and LH_2 tank form the structural backbone of the vehicle and are fabricated from aluminum alloy. The fact that the booster is a heat-sink vehicle dictates internal stiffening to provide a smooth aerodynamic surface. A comparison was made between waffle-pattern stiffening, isogrid stiffening, and stringers with frames to determine the approach which would result in the lightest weight. The results indicated that stringers with frames were lightest, and were selected for these main body components. Since the thermal environment around the vehicle varies, the heat-sink material thickness varies—which leads to tapered skins.

In addition to carrying basic body loads, the LO_2 tank provides support for the forward orbiter attachment through two load-introduction frames within the LO_2 tank. The intertank adapter is a cylindrical section with internal frames and stiffening that transmits body loads between the tanks, supports the canard, and diffuses orbiter drag loads into the booster structural shell.

The LH_2 tank provides the primary structure in that area, supplies the aft orbiter attachment, and supports the wing. The internal frames and stringers are of simple cross section to facilitate the installation of the cryogenic insulation, which is required to prevent cryopumping and to reduce boil-off. Figure 4-47 illustrates the LH_2 tank structural concept. The insulation system uses PPO foam, Lexan plastic caps on the stringers, and balsa blocks.

SD 71-141-1

ALUMINUM
BERYLLIUM
TITANIUM
INCONEL 718
RENE' 41
HS 188

650°
1,650
650
UPPER &
LOWER
UPPER &
LOWER
900
1,650
650
500
1,900
300
TEMPERATURES °F
1,350

Figure 4-46. B-17E Materials and Peak Temperatures

TYP FRAME
DEPTH = 8.00 IN.
STRUCTURE
MATERIAL 2219-T87

36.0 IN.
TYP.

1.0 IN.
(0.3 IN. EFFECTIVE)

0.01 IN. t

2.5 IN.
TYP.

LEXAN
MOLDING

PPO
FOAM

BALSA
BLOCK

ADHESIVE
POLYURETHANE
CREST 7343

Figure 4-47. B-17E LH2 Tank Structure Design

The issue of fracture mechanics is significant with respect to the booster, which has large integral tanks as the primary body structure. An analysis made for the heat-sink LH_2 tank indicated that although the parameter K_C decreased with increasing thickness, the limit design stress values also decreased, resulting in an increase in critical crack length with increasing thickness.

The B-17E heat-sink booster requires two major fairings: a canard fairing and a wing/body fairing. The major problem with these fairings, which are attached to cryogenic tanks, is to accommodate the thermal excursions due to cryogenic contraction and aerodynamic heating. The selected approach for the canard fairing is a beryllium skin operating at a relatively moderate temperature of 500 F. The skin is segmented into panels to handle thermal expansion.

Titanium 6A1-4V (at 850 F) was selected for the wing fairing because it is dependent on the skin for structural integrity. The panels of the wing fairing are attached in a way to allow motion under thermal stress and loading conditions.

The wing structural arrangement is a fail-safe multispar, multirib configuration incorporating open corrugation titanium cover panels on the upper surface and a smooth titanium heat-sink plate stringer lower surface. A hot structure Rene 41 leading edge is used. The canard structure is a spar, rib construction with smooth titanium heat-sink cover panels. The leading edge is made of HS 188. The vertical stabilizer is a titanium heat sink similar to that of the B-9U, with the leading edge made of Rene 41.

4.4.8 PROPULSION AND FLUID SYSTEMS

The propulsion and fluid system concepts of the B-17E are similar to those of the Phase B baseline B-9U. Minor changes are required due to the change in vehicle size, engine thrust level, and the fact that the booster is a heat sink. A heat-sink booster demands that any lines traversing the vehicle must be routed externally within an upper surface fairing.

The main propulsion system consists of thirteen 415,000-pound-thrust engines with an expansion ratio of 25 to 1. The engines are located in the base of the vehicle in a 2 - 4 - 4 - 3 arrangement.

The LO_2 feed system uses two large lines mounted on the vehicle upper surface instead of the four small lines used on the B-9U. The LH_2 feed system has an additional line to feed the thirteenth engine.

The B-17E ACPS has 26 engines, compared with 30 on the B-9U. The total impulse requirement is slightly reduced, which results in a small reduction in propellant tankage.

The air-breathing engine system for the B-17E is similar to that of the B-9U except that the four engines located under the body are eliminated, leaving a total of eight engines. The engines are JTF-22As. A vertical engine deployment is used.

The power systems (APU, hydraulic, and electrical) are similar in concept and configuration to those of the B-9U, with minor changes to account for the reduction in vehicle size and flight time.

The environmental control and life support system is almost identical to that of the B-9U.

The vehicle purge and vent system is still required for B-17E; however, it is significantly reduced due to the elimination of a large part of the TPS. The areas requiring purging and venting are the thrust structure, intertank, and canard and wing fairings.

4.4.9 AVIONICS

The avionics subsystem of the B-17E is essentially the same as that of the Phase B baseline B-9U. Hardware changes would be limited to some reduction in the sensor requirements due to the reduction in vehicle size. There would be somewhat more impact on the avionics subsystem software due to the changes in mission characteristics.

4.4.10 MASS PROPERTIES

The final mass properties for the B-17E booster were based on detailed subsystem analyses. These final weights, as well as the initial weight estimates, are presented in Table 4-11.

Table 4-11. Summary Weight Statement (Launch Condition)

CONFIGURATION B-17E BOOSTER		Initial Estimate		BY GD/C		DATE Revised Weights			
CODE	SYSTEM								
1.0	WING GROUP	49453				51613			
2.0	TAIL GROUP	15487				13231			
3.0	BODY GROUP	179618				174380			
4.0	INDUCED ENVIR PROTECT	—				—			
5.0	LANDING, DOCKING	19856				18942			
6.0	PROPULSION, ASCENT	99012				91921		·	
7.0	PROPULSION, CRUISE	30498				33270			
8.0	PROPULSION, AUXILIARY	6596				10909			
9.0	PRIME POWER	1411				1810			
10.0	ELECTRICAL CONV & DISTR	1600				1600			
11.0	HYDRAULIC CONV & DISTR	2395				1855			
12.0	SURFACE CONTROLS	5702				8788			
13.0	AVIONICS	4600				5400			
14.0	ENVIRONMENTAL CONTROL	1249				1648			
15.0	PERSONNEL PROVISION	·1300				1406	·		
16.0	RANGE SAFETY								
17.0	BALLAST								
18.0	GROWTH	32617				33256			
19.0									
	SUBTOTAL (DRY WT)	451394				450029			
20.0	PERSONNEL	476				476			
21.0	CARGO								
22.0	ORDNANCE								
23.0	RESIDUAL FLUIDS	11534				9577			
24.0									
	SUBTOTAL (INERT WT)	463404				460082			
25.0	RESERVE FLUIDS								
26.0	INFLIGHT LOSSES	14950				17539	·		
27.0	PROPELLANT-ASCENT	2322205				2322497			
28.0	PROPELLANT-CRUISE	58615				58615			
29.0	PROPELLANT-MANEUV/ACS	1100				1360	·		
30.0									
	TOTAL (GROSS WT) LB	2860274				2860093			

DESIGNATIONS:

NOTES & SKETCHES:

Ascent propellants include thrust decay propellants (4708 lb estimated, 5000 lb revised weight), which are not included in performance calculations.

FORM 3945-A-13 NEW 8-70

4.5 OEHT SHUTTLE UPDATE

At the end of the Orbiter External Hydrogen Tank Study activities, a final orbiter external hydrogen tank shuttle sizing was conducted to incorporate the final Orbiter External Hydrogen Tank Study results as well as the most recent Phase B Reusable Shuttle Study results. Significant items from the Phase B study were structure and subsystem weight updates, shortened orbiter design, and reduced integrated vehicle drag coefficient. The weights updates resulted from the final structural and subsystem design/weights analyses. The shortened orbiter resulted from the use of two manipulators to remove the cargo rather than rotating the cargo out, allowing the orbiter body to be be reduced 70 inches. The reduced drag coefficients are from the recent wind tunnel model testing at Ames Research Center on the NR/GD integrated vehicle. Significant items from the Orbiter External Hydrogen Tank Study are increased orbiter weight and reduced tank weight relative to the vehicle sizing results at the beginning of the preliminary design.

The characteristics of this vehicle are summarized in Table 4-12. The sensitivity of this final orbiter external hydrogen tank shuttle vehicle to booster and orbiter weight changes and main engine specific impulse changes are shown in Figure 4-48.
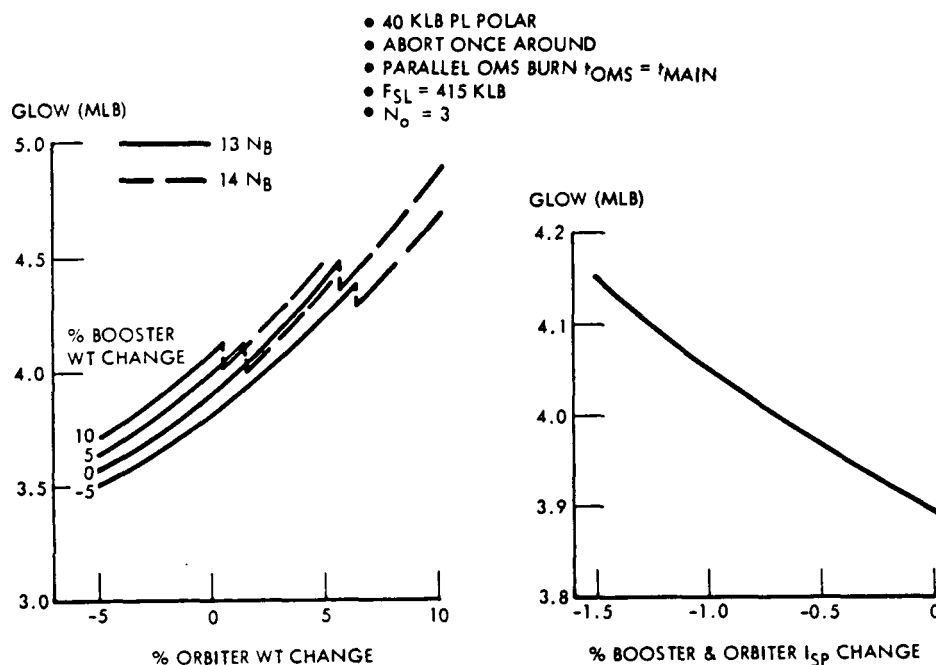

Figure 4-48. OEHT Sensitivity Data

SD 71-141-1

Table 4-12. Final External Orbiter LH$_2$ Tank Configuration Description
(Payload: 40,000 pounds to polar orbit without ABES)

Item	Booster	Orbiter
System GLOW, lb	3,896,070	
T/W at liftoff	1.385	
Stage gross weight, lb	2,724,840	1,171,230
Main propellant usable weight, lb	2,208,618 ①	835,999 ①
OMS propellant weight, lb	-	30,851 ② ③
F/B fuel weight, lb	49,429 ④	-
F/B range, nm	213	-
Stage dry weight, lb	439,867	216,177 ⑤
Tank dry weight, lb	-	22,117
Stage reentry weight, lb	496,622	260,275 ⑥
Stage landing weight, lb	447,389	259,108 ⑥
Max q, psf	573	
Staging, V_r, fps	7333	
h, ft	201,449	
Y_r, deg	16	
q, psi	14.5	
Main engine F_{SL}, lb	415K	
F_{VAC}, lb	455K	477K ⑦
No of main engines	13	3

① Usable propellant based on nominal I_{sp}. Tanks accommodate 1.5-percent additional usable propellant for minimum I_{sp} plus appropriate reserves and residuals.

② Partially filled. Tank is sized to contain propellant for 2000 fps on-orbit ΔV on space station resupply mission.

③ 18,484 pounds are burned during nominal ascent.

④ Tank sized for 56,000-pound capacity.

⑤ Excludes external tank dry weight.

⑥ Includes 40,000-pound payload weight.

⑦ Common power head with booster engine.

4.6 FULLY REUSABLE SHUTTLE UPDATE

While the Orbiter External Hydrogen Tank Study was being conducted, a final reusable vehicle update was being made in the Mainline Phase B Study. This final update incorporated final booster and orbiter weights, reduced integrated vehicle drag coefficient, and improved orbiter design by reducing orbiter length. These results were also incorporated into the final OEHT tank concept update. Utilizing the results of the Phase B baseline studies, a final three-engine orbiter configuration was sized so that the two-engine and three-engine reusable configurations can be compared to each other, as well as compared to the final orbiter external hydrogen tank configuration.

The basic configuration difference between the two-engine baseline 161C orbiter and the two-engine shortened 161C orbiter is in the reduction of the payload bay from 800 inches in length to 730 inches in length and the subsequent reduction of 70 inches in length of the reference body length from 2178 to 2108 inches. The reduction in payload bay is possible because of the elimination of the original design for end-rotating the payload out of the payload bay and the subsequent use of the two manipulators to remove the payload directly (laterally) out of the payload bay. The 730 inches of payload bay length allows 5 inches of clearance to each end of the 60-foot (720 inches) payload. The updated characteristics of the Phase B reusable vehicle, which uses a two-engine orbiter, are shown in Table 4-13.

During the second phase of the study OEHT, the main engine thrust level desired for the fully reusable system employing a three-engine orbiter was established. A 480,000-pound, sea-level thrust engine was selected. Characteristics of the resized, fully reusable vehicle with a three-engine orbiter are shown in Table 4-14.

Table 4-13. Two-Engine Orbiter Fully Reusable Space Shuttle Size
From Phase B Update (Payload: 40,000 pounds to polar
mission with ABES out)

Item	Booster	Orbiter
System GLOW, lb	4,740,357	
T/W at liftoff	1.392	
Stage gross weight, lb	3,861,489	878,868
Main propellant weight, usable, lb	3,108,606 [1]	558,234 [1]
OMS propellant weight, lb	-	33,712 [2][3]
Flyback fuel weight, lb	107,582 [4]	-
Flyback range, nm	354	-
Stage dry weight, lb	607,919	224,169
Stage reentry weight, lb	726,387	270,597 [5]
Stage landing weight, lb	618,805	269,404 [5]
Max q, psf	635	
Staging, V_r, fps	10,773	
h, ft	231,027	
γ_r, deg	6	
q, psf	9.4	
Main engine, F_{SL}, lb	550K	
F_{VAC}, lb	604K	632K [6]
No. of main engines	12	2

(1) Usable propellant based on nominal I_{sp}. Tanks accommodate 1.5-percent additional usable propellant for minimum I_{sp} plus necessary provisions for reserves and residuals.

(2) ΔV loaded is 1500 fps, tank sized for 2000 fps.

(3) 13,892 is burned during nominal ascent assist.

(4) Tank sized for 122,000-pound capacity.

(5) Includes 40,000-pound payload and ABES out.

(6) Common power head with booster engine.

Table 4-14. Final Three-Engine Orbiter Fully Reusable Space Shuttle
Size (Payload: 25,000-pounds to logistic resupply
mission with ABES in)

Item	Booster	Orbiter
System GLOW, lb	4,479,313	
T/W at liftoff	1.29	
Stage gross weight, lb	3,113,577	1,365,736
Main propellant usable weight, lb	2,549,769 (1)	976,874 (1)
OMS propellant weight, lb	-	37,074 (2) (3)
Flyback fuel weight, lb	53,255 (4)	4,342
Flyback range, nm	189	
Stage dry weight, lb	480,326	292,626
Stage reentry weight, lb	542,003	330,663 (5)
Stage landing weight, lb	488,748	324,939 (5)
Max q, psf	509	
Staging, V_r, fps	7235	
h, ft	196,368	
Y_r, deg	14	
q, psf	17.1	
Main engine, F_{SL}, lb	480,000	
F_{VAC}, lb	526,000	551,000 (6)
No. of main engines	12	3

(1) Usable propellant based on nominal I_{SP}. Tanks accommodate
1.5-percent additional usable propellant for minimum I_{sp} plus
necessary provisions for reserves and residuals.

(2) ΔV loaded is 1500 fps, tank sized for 2000 fps.

(3) No OMS ascent assist.

(4) Tank sized for 55,000 pound capacity.

(5) Includes 25,000-pound payload and ABES IN.

(6) Common power head with booster engine.

4.7 MANUFACTURING REQUIREMENTS

This section presents the orbiter, tanks, and booster manufacturing requirements.

4.7.1 ORBITER

Evaluation of candidate orbiter vehicles during Phase 1 of the orbiter external hydrogen tank vehicle study included a manufacturing assessment of the vehicles, which was done by reviewing all conceptual layout drawings of vehicle configuration and selecting items of significant design differences for evaluation.

A subjective rating system was used to compare the manufacturability and cost of each area of difference. When selection of the top four vehicles was made, manufacturing procedures, tooling, rough order of magnitude cost, and schedule data were defined to support the orbiter selection (Volume II, 4.4.10).

The final configuration was firmed up, and subsystem definition drawings were evaluated to develop a supplemental plan for comparison against the current baseline. The plan includes changes to manufacturing procedures, facilities requirements, tooling, and quality control, and reflects changes or new manufacturing problems (Volume II, 6.1 and Volume III, 6.1).

The OEHT configuration of the orbiter is so much like the reusable orbiter that changes in the manufacturing operations are minimal. Table 4-15 shows the impact of the OEHT configuration on the orbiter manufacturing operation. Design changes are required in the detailed fabrication tooling. Time is saved because of the smaller, simpler forward tank (LO_2 instead of LH_2). This reduction in forward tank assembly time is offset by additional time required for REI installation, increased time for installation of propulsion systems, and increased checkout time for the added systems.

4.7.2 TANK

Several external hydrogen tank configurations were considered and rated along with various ways of insulating them (see Volume II, 4.5.9).

After final selection of configuration and after the definition of structural and subsystems design, a manufacturing plan was developed for

- 105 -

Table 4-15. Impact to the Orbiter Manufacturing Plan

Design Differences	Tooling Impact	Build Sequence Changes	Fabrication/Assembly Time	Revised Procedures	Schedule Changes
Forward fuselage (shorter)	No	No	Less (less REI)	No	Yes, (shorter time span)
Forward tank (LO$_2$ instead of LH$_2$ - 50% less volume)	Yes (added kick frames and internal systems provisions; additional welds; additional machines; T$_1$ fittings on aft skirt)	Yes	Less (no cryogenic insulation)	Yes	Yes, (eight weeks shorter)
Intermediate fuselage (more TPS; tank arrangement changes)	Yes (additional tooling required at external tank umbilical and attachment fittings; added doors)	No	Greater (more REI; more propulsion system; removal of main LO$_2$ tanks)	No	Yes, (longer time span)
Aft fuselage (three engines instead of two; tanks relocated)	Yes (more tooling)	No	Greater (more systems; additional engine to checkout)	No	Yes, (longer time span)
Vertical Stabilizer (smaller	No	No	Less (smaller; less REI)	No	Yes, (slightly shorter time span)
Wing (more TPS; no LO$_2$ feed line through wing)	No	No	Greater (more REI on wing upper side)	No	Yes, (longer time span)

fabrication, and changes to the facility utilization and manufacturing plan (Volume III, 6. 1, Orbiter, SD71-104-1) were prepared.

The tank build sequence is shown in Figure 4-49. The bulkheads (Figure 4-50) are bulge-formed in parallel with the cylinder assembly (Figure 4-51). They are welded together in a horizontal tool (Figure 4-52). Then the tank is inspected, cleaned, closed out, pressurized, and lifted into a vertical tool where the SOFI is applied (Figure 4-53). The tank is moved to the machining station for final trimming of the SOFI. At another vertical station, the cork is applied (Figure 4-54) and then the tank is placed on a trailer for removal to the storage building (Figure 4-55).

The design of the external LH_2 tanks was conceived with engineering, manufacturing, and program requirements in mind. Ease of manufacturing, accessibility to equipment, and reduced costs were considered. The retainment of equipment in the orbiter rather than in the ejected tank made possible the reuse of the equipment. The increase in the gauge of part of the tank skins made possible the handling of the unpressurized tank assembly during manufacturing and on the orbiter (unfueled) in either the horizontal or vertical position. A listing of the design features for the reduction in program costs is as follows.

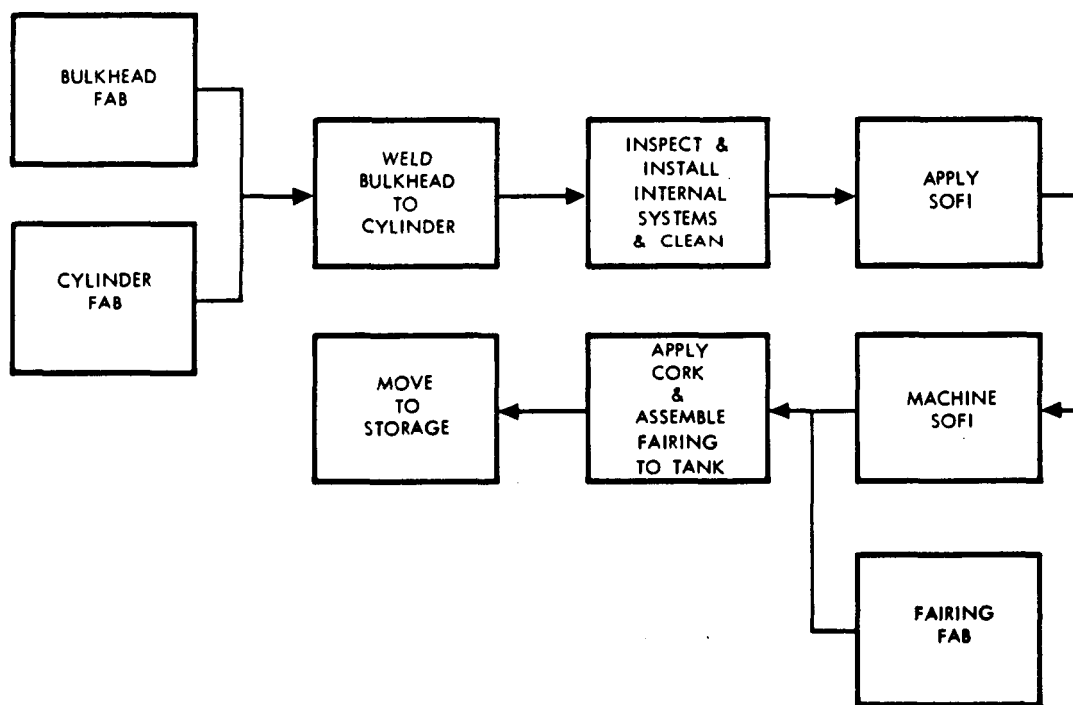1. SOFI on the exterior of tank rather than in the interior

Figure 4-49. External LH_2 Tank Build Sequence

STANDARD WELD SEQUENCE
O POSITION
O CLAMP
O TRIM (FOR FIT)
O CLEAN
O REPOSITION
O WELD
O MACHINE OFF WELD BEADS

TYPICAL DYE PENETRANT INSPECTION SEQUENCE

Weld area to be submitted to inspection in a clean condition suitable for penetrant application.

1. Spray penetrant on weld area utilizing the portable airless spray control unit.

2. After minimum period for penetration, remove surface penetrant by spray washing with deionized water. Dry surface with hot air.

3. Spray developer on weld area utilizing the portable airless spray control unit.

4. After minimum period for developer activation, place portable darkroom in position over weld area and energize the black lights.

5. Inspect for weld defects and mark area.

6. Remove all developer and residual penetrant with deionized water; spray and brush utilizing portable spray unit.

7. Dry all surfaces using hot air and inspect with black light to assure removal of all penetrant materials.

TYPICAL RADIOGRAPHIC INSPECTION SEQUENCE

1. Install x-ray marker tape adjacent to weld bead down entire length of weld.

2. Load x-ray film in film reel unit and mount reel carriage on track.

3. Mount x-ray tube and carriage on track.

4. Energize x-ray radiation warning system.

5. Activate x-ray unit and initiate automatic sequencing operations required to traverse entire length of weld.

6. Remove x-ray film reel unit from track and deliver to x-ray film processing area for developing.

7. Interpret x-ray film and mark weld defects.

8. Locate and mark defects on weld.

9. Remove x-ray tube and carriage from track when weld is free of defects requiring additional x-rays.
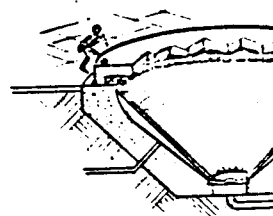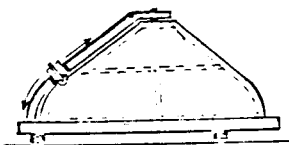
10. Remove x-ray marker tape.

EAD FAB →
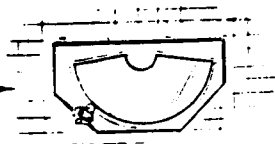
○ FLAT SHEETS 30' X 19' 2014-0 COND
○ CUT TO CONE SHAPE PATTERN
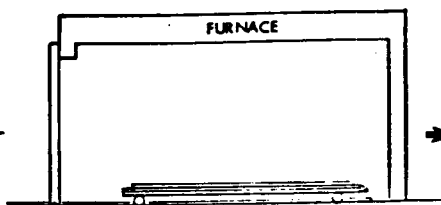
FURNACE

○ PLACE FLAT SHEET(HALF BLKD SEC.) IN FURNACE
○ HEAT TO 925° – THEN QUENCH
○ MATERIAL NOW 2014T4 ALUM.

○ PLACE IN BULGE FORM TOC

○ PLACE ON WELD TOOL, CLAMP IN PLACE
○ PERFORM WELD SEQUENCE (FIRST WELD)
○ MOVE WELD PACK TO OPPOSITE SIDE, AND CLAMP
○ MACHINE OFF WELD BEADS
○ INSP. PERFORM DYE PEN SEQ.
○ INSP. PERFORM X-RAY (MFG. LEAVES STATION FOR 1 HR)
○ PERFORM WELD SEQUENCE (SECOND WELD)
○ MOVE WELD PACK, AND MACHINE OFF WELD BEADS
○ INSP. PERFORM DYE PEN. SEQ.
○ INSP. PERFORM X-RAY (MFG. LEAVES STATION FOR 1 HR)
○ INSP. VERIFY X-RAY
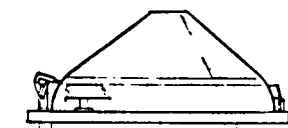
○ SET FORWARD BLKD (ONLY) ON POSITIONER
○ CUT OUT 12" SEGMENT

○ SEND BLKDS TO VENDOR FOR CHEM MILL
OF WELD LANDS AND FINAL SHAPING
(THICKNESS)
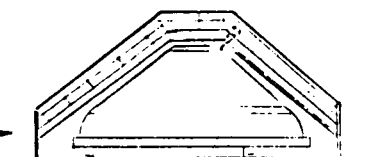
○ FORGING OF ACCESS COVER AND DOLLAR RING
(VENDOR SUPPLIED)
○ POSITION DOLLAR RING TO BLKD AND CLAMP IN PLACE
○ PERFORM WELD SEQUENCE
○ MACHINE OFF WELD BEADS
○ PERFORM DYE PEN SEQ.
○ PERFORM X-RAY
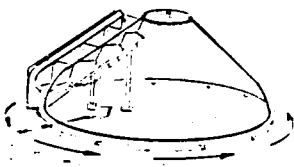
○ MOVE BLKD TO CLEANING STATION
○ SOLUTION CLEAN
○ SPRAY ON M602 PRIMER
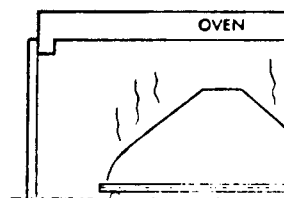
OVEN

○ OVEN CURE (290° – 300°F 2-

PANEL FABRICATION →
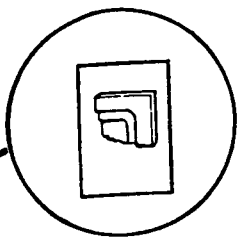
○ FLAT SHEETS RECEIVED 45' x 19' 2014T6 ALUM
○ SEND MATERIAL TO VENDOR FOR CHEM MILLING OF WELD LANDS
○ RECEIVE PANELS AND TRIM TO ROUGH SIZE

○ CLEAN PANELS
○ SPRAY ON M60

O HEAT TREAT SEG. TO 2014T6 CONDITION
(OVEN 310 - 330°F 19 HRS)

L AND SHAPE BLKD.HALF SEGMENT

O PLACE BOLTING RING SEGMENTS
(180°) INTO WELD TOOL
O PERFORM WELD SEQUENCE

O PLACE FWD BLKD IN UNDER SLUNG POSITIONER FOR WELDING
O CLAMP IN PLACE AND PERFORM WELD SEQUENCE
O REPEAT FOR SECOND WELD

FAIRING ATTACH RING

-3 HRS)

O BULKHEAD IS COMPLETE

O CURE IN OVEN (290° - 300°F 2-3HRS)

2 PRIMER (BOTH SIDES)

Figure 4-50.   Bulkhead Fabrication

CYL. FAB

END VIEW
INSPECTION PLATFORM
DYE PEN. OPERATION
X-RAY OPERATION

MONITOR SCREEN
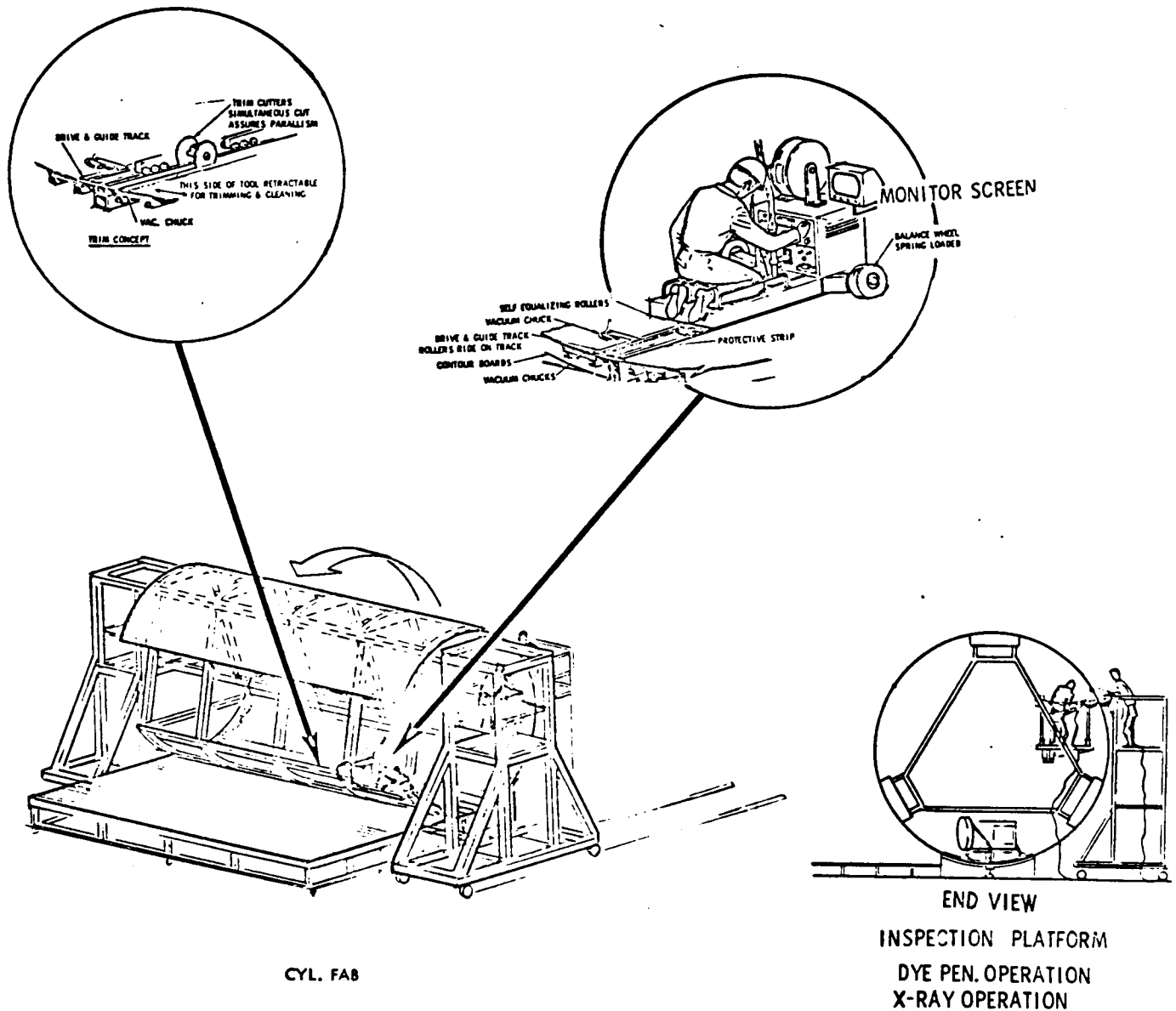
O POSITION FIRST PANEL ON TOOL AND CLAMP IN PLACE (VACUUM CUPS)
O SET SECOND PANEL ON TABLE OF TOOL AND CLAMP IN PLACE (VACUUM CUPS)
O PERFORM WELD SEQUENCE (FIRST WELD)
O ROTATE PANELS AND POSITION THIRD PANEL ON TABLE, CLAMP IN PLACE
O MACHINE WELD BEADS OFF ON FIRST WELD
O INSPECTION PERFORM DYE PEN SEQ. ON FIRST WELD (WHILE SET UP IS MADE FOR SECOND WELD)
O INSPECTION PERFORMS X-RAY (MFG. LEAVES AREA FOR 1 HR)
O PERFORM WELD SEQUENCE (SECOND WELD)
O ROTATE PANELS AROUND FORMING COMPLETE CYL.
O TRIM FOR SIZE, AND CLAMP FOR CLOSEOUT WELD
O MACHINE WELD BEADS OFF ON SECOND WELD
O INSPECTION PERFORM DYE PEN SEQ. ON SECOND WELD
O INSPECTION PERFORMS X-RAY (MFG. LEAVES AREA FOR 1 HR)
O PERFORM WELD SEQUENCE (THIRD CLOSEOUT WELD)
O ROTATE CYL. TO INSP. POSITION AND MACHINE OFF WELD BEADS
O INSPECTION PERFORMS DYE PEN. SEQ. ON THIRD WELD
O INSPECTION PERFORMS X-RAY (MFG. LEAVES AREA 1 HR)
O WHILE ALL X-RAYS ARE BEING READ FOR DEFECTS, MFG. APPLIES TEMPLATE AND INSTALL
   STUDS ON CYL. (PERCUSSION STUD WELDING)

Figure 4-51. Cylinder Assembly

2. Monocoque tank structure pressure-stabilized rather than stiffened

3. Simple Chem-Mill pattern for weld lands

4. Cylinders (fabricated of three maximum size aluminum sheets rather than of many conventional size sheets welded together

5. Elimination of PU system, use of sensor installation for fill

6. Tank and reaction for load inputs

WELD TOOL
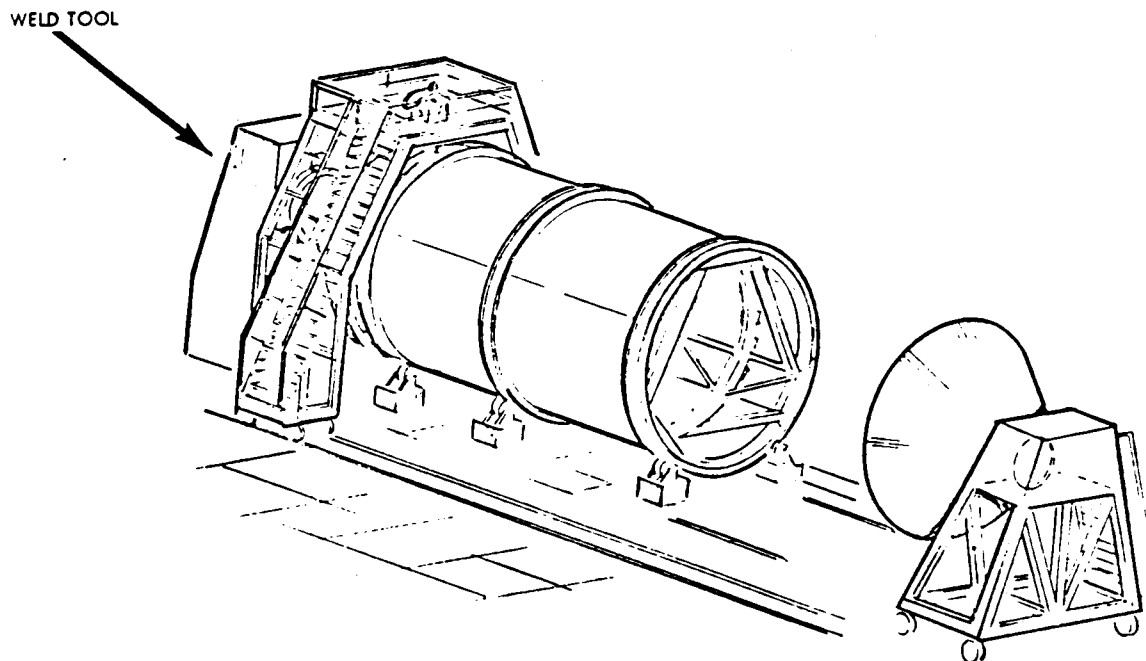
O POSITION BLKDS ON TOOL
O WELD PERCUSSION STUDS ON EACH BLKD
O POSITION CYL. IN TOOL AND INSTALL SIZING RINGS FOR BLKD TO CYL. CLAMPING
O PERFORM WELD SEQ. (ROTATING PART, WELD EQUIP. STATIONARY)
O MACHINE OFF WELD BEADS
O POSITION SECOND BLKD TO CYL.
O PERFORM DYE PEN SEQ.
O PERFORM X-RAY (MFG. LEAVES AREA 1 HR)
O PERFORM X-RAY SEQ. (MFG. LEAVES AREA)
O WHILE X-RAYS ARE BEING VERIFIED MFG. WILL APPLY ROOM CURE PRIME (BRUSH ON) TO STUD WELD PADS
O INSTALL PRESSURE LINE SEGMENTS TO STUDS (INSIDE TANK)

Figure 4-52. Welding of Bulkheads to Cylinder

SPRAY ON FOAM OPERATION

START SPRAY OPERATION

PROGRESSIVE SPRAY

TRIGGER OPERATION OF NEXT LEVEL

PROGRESSIVE SPRAY

PROGRESSIVE SPRAY

STOP SPRAY
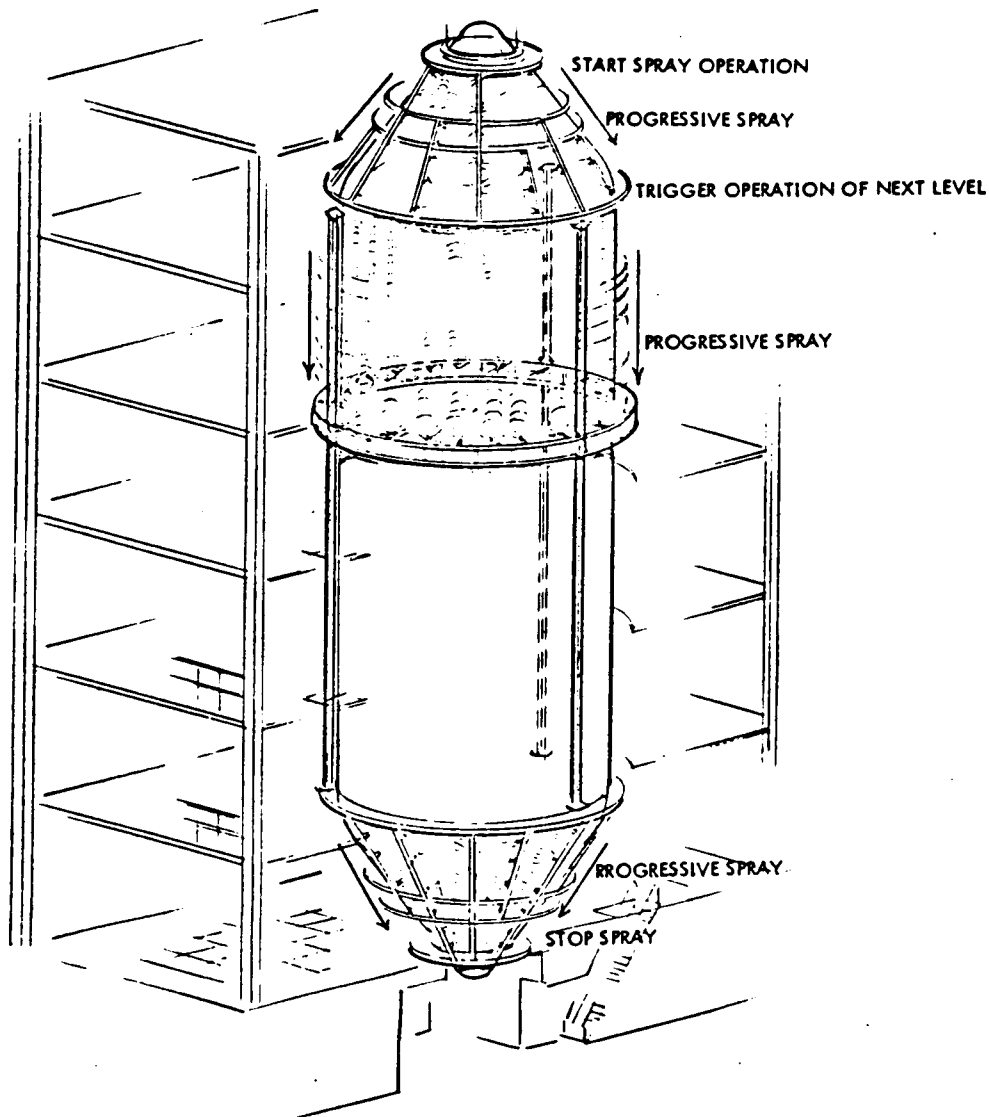
O PREPARE TO SPRAY ON INSULATION
O SPRAY OF SOFI (ONE SINGLE APPLICATION, NO CLOSE OUT SEALS REQUIRED)
O MOVE TO SPRAY FOAM MACHINING STATION

Figure 4-53. Application of SOFI

SD 71-141-1

SYSTEM INSTALLATION
O REMOVE MATERIAL HANDLING EQUIPMENT
O LOWER FAIRING OVER FORWARD BULKHEAD
O ALIGN AND BOLT
O REMOVE FWD FAIRING CONE AND ATTACH STRUTS AND HOOK UP VENT LINE AND HARNESSES
  (CHECK ELECTRICAL SYSTEM)
O REPLACE FWD FAIRING CONE
O INSTALL ABLATOR STRIPS, CLOSE OUT FAIRING
  ATTACH BOLTS AT BULKHEAD, DO NOT PUT ABLATOR PLUGS IN ON CONE
O ATTACH MATERIAL HANDLING EQUIPMENT TO FAIRING
O LIFT TANK SLIGHTLY AND DEPRESSURIZE
O REMOVE BLANKING PLATES AND INSTALL
  FEED AND PRESSURE LINES, AND LOW PRESSURE WARNING SYSTEM
O PRESSURIZE TO 2 PSI, AND PERFORM LEAK CHECK
O CONTINUE TO LIFT AND REMOVE FROM STATION
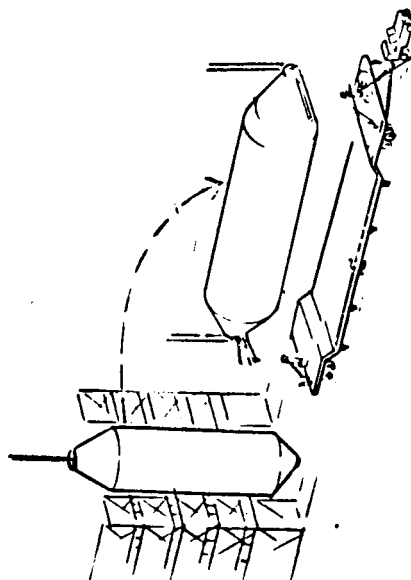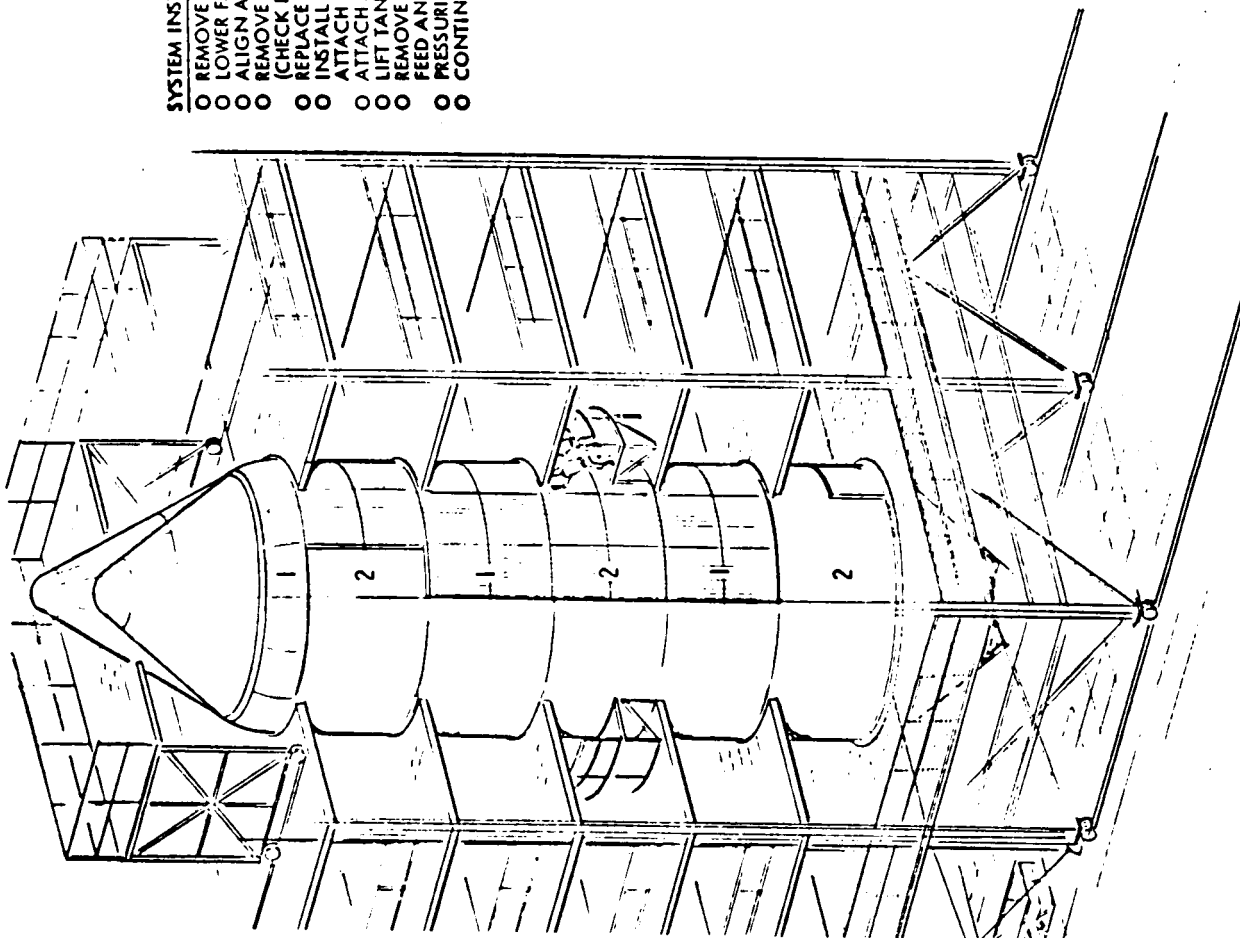
O ROTATE TO HORIZONTAL POSITION AND LAY ON SHIPPING DOLLY
O CLOSEOUT PAPERWORK, DD250

Figure 4-54. Cork Installation

STATIONS

1. INSP. & CLEANING
2. INSP. & CLEANING
3. INSP. & CLEANING
4. WELD REPAIR
5. SOFI APPLICATION
6. SOFI MACHINE
7. CORK APPLICATION
8. CORK APPLICATION

RESTROOMS
OFFICES
SOFI CLEANING
& STORAGE AREA

← ELEVATORS

← RESTROOMS
← X-RAY PROCESSING ROOM
← OFFICES

VERTICAL BUILDING
125' x 175' x 110' high

VERTICAL ASSEMBLY BUILDING
← APPROX 5 MI →

LOWBOY TRANSPORTER
GUIDE TRACK
AIR GLIDE PADS
HYDRAULIC LIFT
STANCHION

TANK STORAGE BUILDINGS

O SHIP TO STORAGE AREA
O REMOVE FROM DOLLY AND PLACE IN STORAGE ON CONCRETE STANCHIONS

Figure 4-55.   Assembly and Storage Buildings

7. Constant weld land thickness

8. Common access manhole diameter

9. Common front and aft bulkhead shapes, use of the same forming tool

10. Increased skin gauge (040 to 080) in spherical area of the bulkheads (handling either vertical or horizontal in unpressurized state)
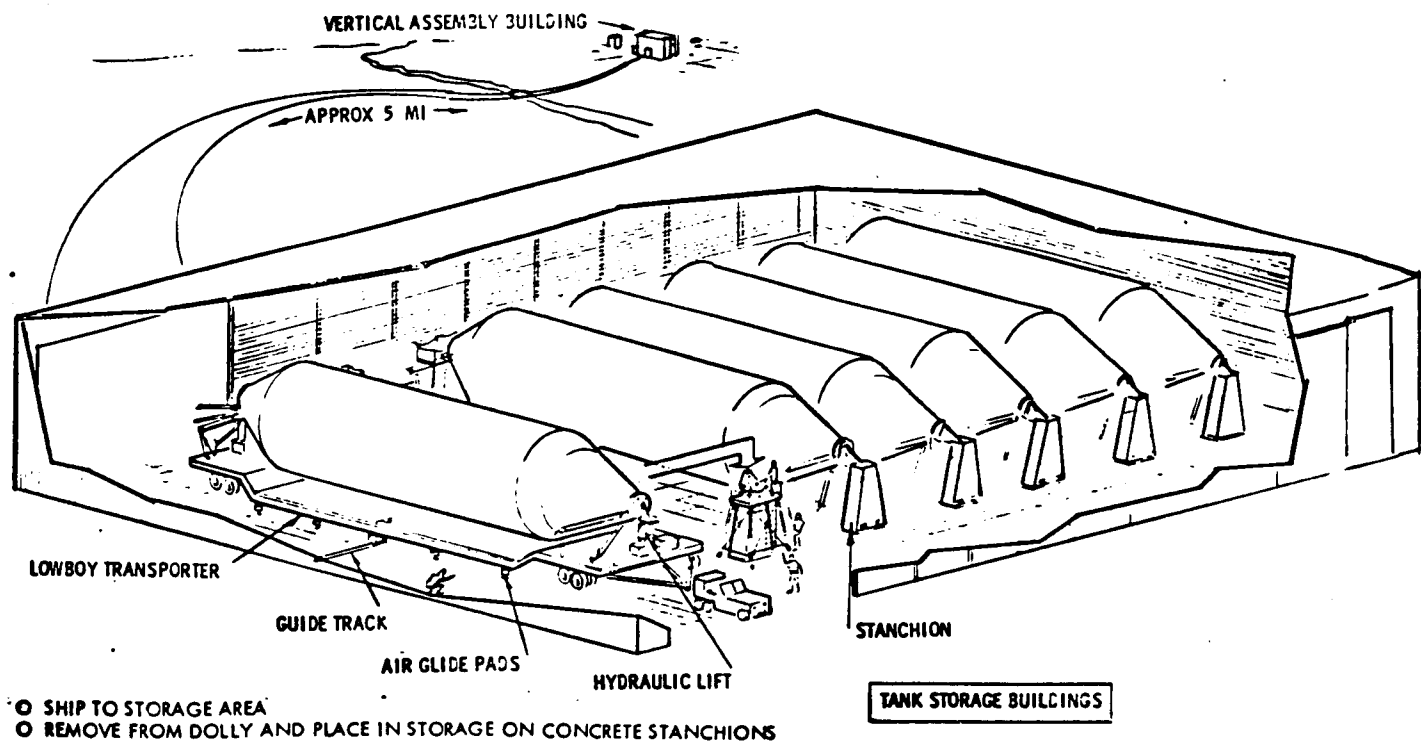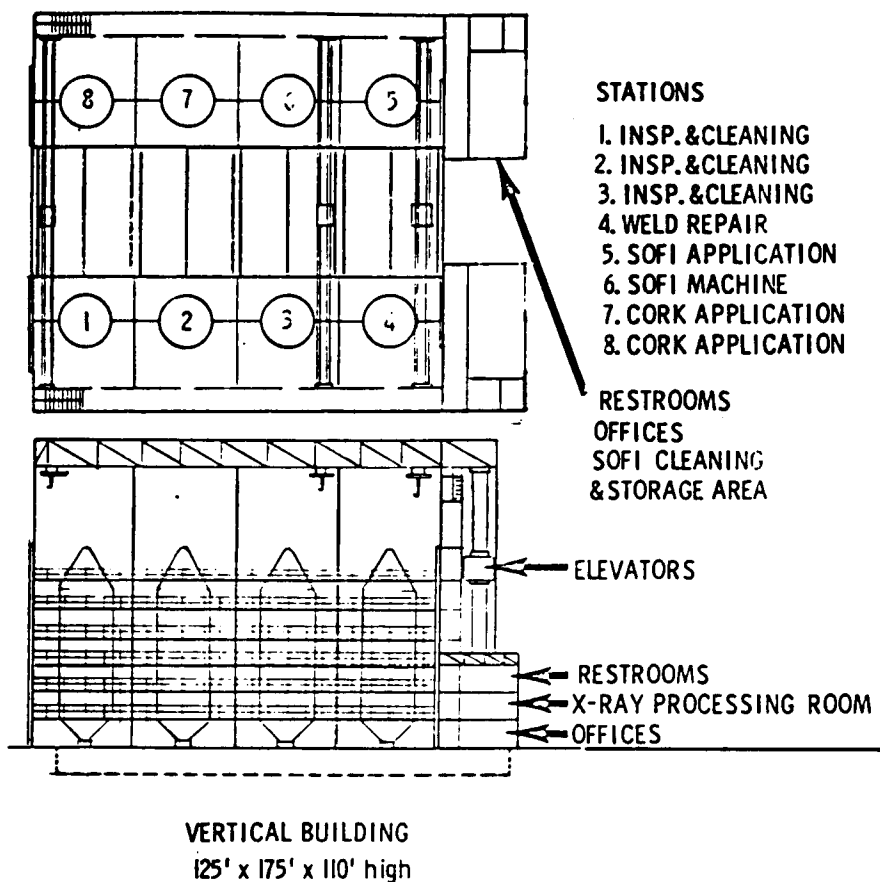
11. Release, attachment, and ejection mechanisms in orbiter (reused),

12. Front end of forward fairing removable for access to installed equipment

13. Vent valve system installed in orbiter (reuse)

14. Self-sealing (not motor driven) disconnect on tank side

4.7.3 BOOSTER

The manufacturing requirements for the B-17E heat sink booster were established. In addition to the overall reduction in vehicle size the major impact on manufacturing occurred with the LO_2 tank, the intertank adapter, the LH_2 tank, and the canard and wing surfaces. Since the body components are heat sink, they require internally stiffened skins and internal frames and bulkheads; in addition, they have skins of variable thickness. This fact will make the manufacturing and assembly operations for these components somewhat more complex than would components of equivalent size which have external stiffening and frames and skin of uniform thickness. The use of a dual insulation scheme of the LH_2 tank and the fact that the interior wall is not smooth will increase the task of installing the insulation.

The wing lower surface and canard surfaces will be titanium skin stringer heat sinks rather than insulated surfaces with a TPS: their manufacture will thus be simplified.

The elimination of a large portion of the TPS results in a significant reduction in the TPS manufacturing task. The two components on the B-17E which could be considered similar to the TPS are the canard and wing fairings. The canard fairings uses beryllium panels; the wing fairing uses titanium. The elimination of the high temperature materials used on the Phase B baseline TPS will simplify the manufacturing of these two sections.

The major manufacturing facilities selected for the Phase B baseline are applicable to the B-17E; however, there would be some change in tooling, fixtures, and facility modifications.

4.8 TEST (INTEGRATED SYSTEMS TEST)

During the initial phase of the study (Part I), an assessment of the candidate external LH_2 tank and orbiter vehicle configurations was made to determine differences in test program requirements. In addition to consideration of the development program, consideration was given also to those requirements associated with ground checkout of the tank and tank/orbiter configurations. The comparative analysis resulted in a qualitative rating of the alternate concepts. It was generally concluded that from a program standpoint, there were no major test differences in the external LH_2 tank configurations.

Part II of the study activity was devoted to establishing the test program impact of using external LH_2 tanks on the orbiter and also the impact of using a heat sink booster. Major additional test activities are associated with the orbiter external tank and the test flow logic for the tank is presented in Figure 4-56. It was concluded that ground tests would provide sufficient confidence to allow the initial integrated vehicle all-up demonstration to be accomplished on the first manned orbital flight. As a result, unmanned flight tests with subscale or full-scale tanks are not required. The results of the test planning effort, including orbiter/tank and orbiter/booster integration tests, are presented in Volumes II and III.
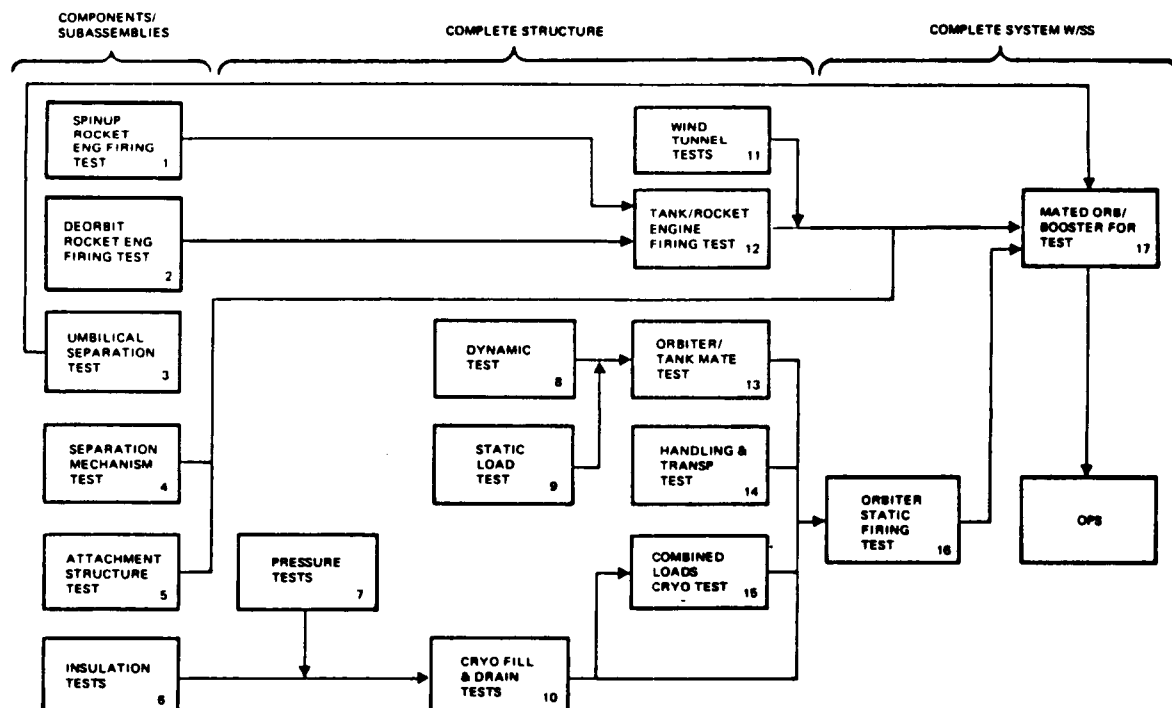

Figure 4-56. Tank Development/Qualification Test Flow Logic

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The impact of the orbiter external hydrogen tank and heat sink booster on the baseline all reusable vehicle test program is summarized in Table 4-16. The table contains testing over and above that required to develop the tanks themselves. The blocked-in areas in the table are the most significant additions to the baseline test program. They include tests to establish the following: (1) the influence of the increased interference heating between the tank and the orbiter and the changed aerodynamic coefficients, (2) the aerodynamic characteristics and the dynamics of the orbiter and tanks during early separation and the identification of the tanks at the time of separation, and (3) the establishment of increased structural loading imposed by vibration testing in a 1-g environment instead of the flight environment of zero-g.

BOOSTER

The test requirements for the B-17E heat sink booster were established. The general test philosophy and test program defined for the Phase B baseline, B-9U, are applicable. The B-17E, a smaller booster, will tend to reduce the test facility size requirements. The reduced aerodynamic heating environment associated with the lower staging velocity will reduce the heating requirements for elevated temperature testing on all components except the LO_2 tank, intertank adapter, and LH_2 tank. These body components, which on B-9U were protected to 250F by a TPS, would be operating at 300F on the heat sink booster and hence have slightly increased testing requirements. The elimination of most of the high temperature TPS will have a significant reduction on the amount of TPS testing. Such testing will be limited to the canard and wing fairing areas.

The fact that the forward orbiter attachment is located in the LO_2 tank will require that the separation system tests make use of an LO_2 tank which is a change from the baseline which used only the intertank and LH_2 tank.

In summary, the overall test program for the B-17E booster will be simpler than for the B-9U.

Table 4-16. Increased Test Requirements

Test	Orbiter Only	Orbiter and Tank	Booster	Integrated Vehicle	Comments
Wind tunnel data required	Aerodynamic coefficient ✓ Heating data ✓	Interference aerodynamic coefficient X Heating X	Aerodynamic coefficient ✓ Heating ✓	Aerodynamic coefficient X Heating ✓	•Similar to baseline •Extended for tank aerodynamic and heating effects •Lower booster staging velocity
Flight environment	M ✓ α ✓ h ✓	M ✓ α ✓ h ✓	M ✓ α ✓ h ✓ Reduced Environment	•Early abort – tank separation problem X	
Dynamic testing with and without external tanks	Horizontal attitude ✓	Horizontal attitude X Loading critical with LH₂ in tanks	Horizontal attitude ✓	Horizontal attitude X Loading critical with LH₂ in tanks	•Consider reduced vibration criteria for loaded LH₂
MPS static firing	With LH₂ loaded ✓	With LH₂ loaded ✓	No change ✓	No change ✓	•No additional testing required ✓
Combined subsystems	No change ✓	•Similar mission sequence ✓ •ASIL ✓	No change ✓	No change ✓	•Small increase in testing ✓

Legend

✓ Similar to reusable orbiter tests.

X Significant change for OEHT.

4.9 OPERATIONS

The external hydrogen tank configuration will create only minor differences from the planned Phase B turnaround activities. A discussion of these differences is contained in the following paragraphs starting with safing area operations and proceeding through launch and flight operations (Figure 4-57).

Safing operations required for the orbiter can be reduced in time because of the elimination of the H_2 tank and the necessity of purge. The time saving in the safing area will be approximately two and one-half hours. No changes in supporting equipment or facilities are contemplated because of other purge requirements.

The difference in the turnaround time for the orbiter external hydrogen tank concept is created by eliminating the vehicle hydrogen tank internal inspection, which is estimated as requiring approximately 14 hours. Once again, the requirement to inspect other parts of the system–in this case the LO_2 tank–will not allow deletion of equipment. There is no planned scheduled maintenance on the H_2 internal tank configuration; thus the serial time for maintenance would be equal. The external H_2 tanks will be installed just
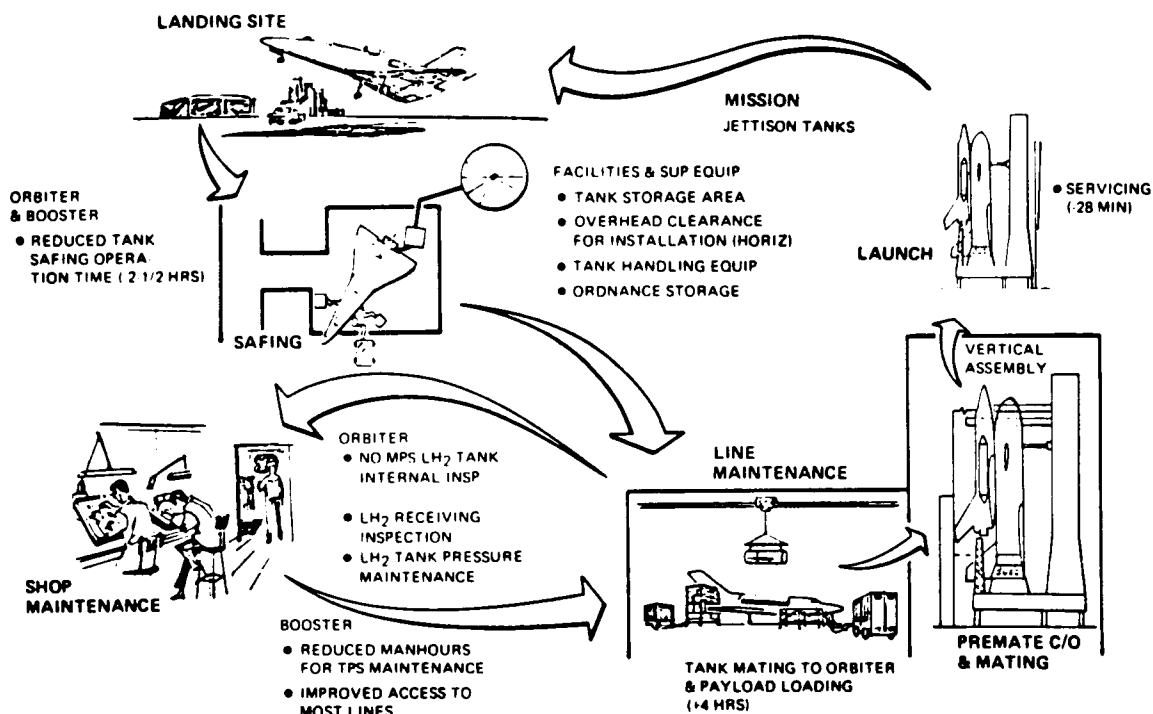
Figure 4-57. Integrated Vehicle Operations and Launch

prior to orbiter premate checkout at the start of the assembly launch phase. It is planned to bring the tanks into the premate checkout station by overhead crane, lower them into position, and attach them in parallel operations. The time allocated is approximately four hours; included is the structural attachment of the tanks and tank/vehicle subsystem interfaces. Functional checkout of the interfaces will be accomplished during the vehicle checkout operations which are already a part of the baseline sequence. Mating/ erecting movement and launch pad operations will be the same as for the baseline. The increase in propellant quantities in the orbiter is more than offset by decreases in the booster; the total requirement is less.

In-flight operations of the orbiter will require jettison and deorbit of the H_2 tanks after the vehicle is in the coast-to-apogee phase. Once the tanks have been jettisoned and the vehicle attitude stabilized, all remaining operations will be the same.

Additional facilities will be required at the operational site to store a sufficient quantity of tanks to assure smooth flow of tanks into the maintenance cycle. Checkout of the tanks at the operational site will be limited to a functional check upon receival, monitoring of an inerting pressure during storage, and a functional check of the end-to-end system after installation.

Support equipment will be required to hoist the tanks into the horizontal and vertical planes, to provide storage pallets, to provide pressurization and monitoring capability, to provide ordnance storage and checkout, and to provide a transportation capability.

Some differences in the purge operations are described in Table 4-17 and in Figure 4-58. The facilities required for purge operations are unchanged since the flow rates which determine the facility sizing are the same as for the reusable vehicles.

Table 4-17. Comparison of Purge Factors

Description	Fully Reusable Orbiter	External Hydrogen Tank Orbiter
Concentrations after purge	1% GH_2, GN_2, GO_2 20% GO_2 (LO_2 tank) Dew point -65 F	Same
Purge media	Media To purge GN_2 Air, GH_2, GOX GH_2 GN_2 for pre-tanking Air GN_2 for pre-tank entry	Same
Tanks and lines purge	H_2 tank purged at maintenance repair area, launch pad, and safing area; O_2 tank purged at safing area	Same, except H_2 tanks not present after mission for safing area purge
Vehicle compartments purge	7090 SCFM total	4090 SCFM total
Compartment vents	Approximately 235 vents Approximately 3000 in.^{2} vent area	Same

PURGE GAS	PRELAUNCH PURGING			POST LANDING SAFING			GROUND MAINTENANCE PROTECTION		
	LH₂ TANKS	LO₂ TANKS	CAVITIES	LH₂ TANKS	LO₂ TANKS	CAVITIES	LH₂ TANKS	LO₂ TANKS	CAVITIES
DEHUMIDIFYING AIR	✓	✓					✓	✓	✓
DRY NITROGEN	✓ ①		✓	⊗②	✓ ③	✓			
GAS (H₂ OR O₂)	✓								
LIQUID (H₂ OR O₂)	✓	✓							

● MAIN PROPELLANT TANKS
● OMS PROPELLANT TANKS
● CAVITIES

DEHUMIDIFYING LEVEL DEW POINT 65 F
① SAFE O₂ LEVEL 1% BY VOLUME
② SAFE H₂ LEVEL 1% BY VOLUME
③ SAFE O₂ LEVEL 20% BY VOLUME
⊗ DELETED FOR EXTERNAL LH₂ TANK ONLY

EXTERNAL TANK TO BASELINE ORBITER DIFFERENCES
● EXTERNAL LH₂ TANKS
● LO₂ TANK FWD
● SMALLER VEHICLE
● LARGER TANKS
● EMPTY MID-BODY

FLOW RATES (PRELAUNCH)
161C BASELINE ORBITER
● ~ 44,000 LB/HR
0500 EXTERNAL TANK ORBITER
● ~ 35,000 LB/HR

EXCLUDES CARGO BAY
● FLOW RATE DEPENDS ON CARGO

Figure 4-58. Purge Requirements Comparison,
External Tank Orbiter

4.10 SCHEDULE SUMMARY

The overall effect of the OEHT shuttle configuration on the 270-day baseline schedules is comparatively small. The orbiter vehicle fabrication time span remains unchanged at 32 months, and the production interval is still established by double-conical tank tooling usage. The only schedule adjustments required involve the subsystem schedules beginning with the Orbiter Program Level 3 schedule. The sequence of activity for these must be started two months earlier to allow time for installation of a net 20 percent more panels of reusable external insulation during orbiter vehicle assembly. Although TPS requirements are reduced by the shorter orbiter, additional TPS is required to protect fuselage and wing surfaces subject to interference heating caused by the externally mounted hydrogen tanks.

The schedule developed for the expendable LH_2 tank starts fabrication of the first tank in January 1975 in support of a scheduled fit-check on the first orbiter vehicle six months later. Fabrication of tanks 2 and 3 follows in support of the orbiter main propulsion test article. Production LH_2 tank fabrication commences in January 1976. The optimum manufacturing build rate supports the present traffic model need dates for the 445 flights although storage of as many as 80 tanks at any one time is required.

4.11 COSTS

4.11.1 TOTAL PROGRAM FUNDING

The total estimated cost, excluding contractor fee, for the orbiter external hydrogen tank configuration for the space shuttle program elements at WBS Level 3 are summarized in Table 4-18. Subsystem costs at Level 5 were identified by hardware on manpower effort, segregated into significant categories of effort (total DDT&E, production, and operations), and then summarized to the total program cost level. The table also shows the theoretical first unit (TFU) cost for both the orbiter and booster.

Figure 4-59 reflects the estimated time phasing of the total shuttle program expenditures for the 17-year period beginning with the Phase C/D authority to proceed (ATP) and continuing through a 10-year operations program. The data presented are the result of summarizing the GFY time-phased work breakdown structure items from Detail Level 5 to the program level. The techniques and rationale for accomplishing this time phasing, together with the detailed tabular data, are the same as for the baseline Shuttle Program (refer to SD 71-107).

GFY	1972	1973	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989
ANNUAL	03	50	1 11	1 68	2 00	1 68	1 21	52	14	09	09	09	10	11	11	11	.08	
CUMULATIVE	03	53	1 64	3 33	5 33	7 00	8 21	8 74	8 87	8 96	9 05	9 14	9 24	9 35	9 46	9 56	9 64	

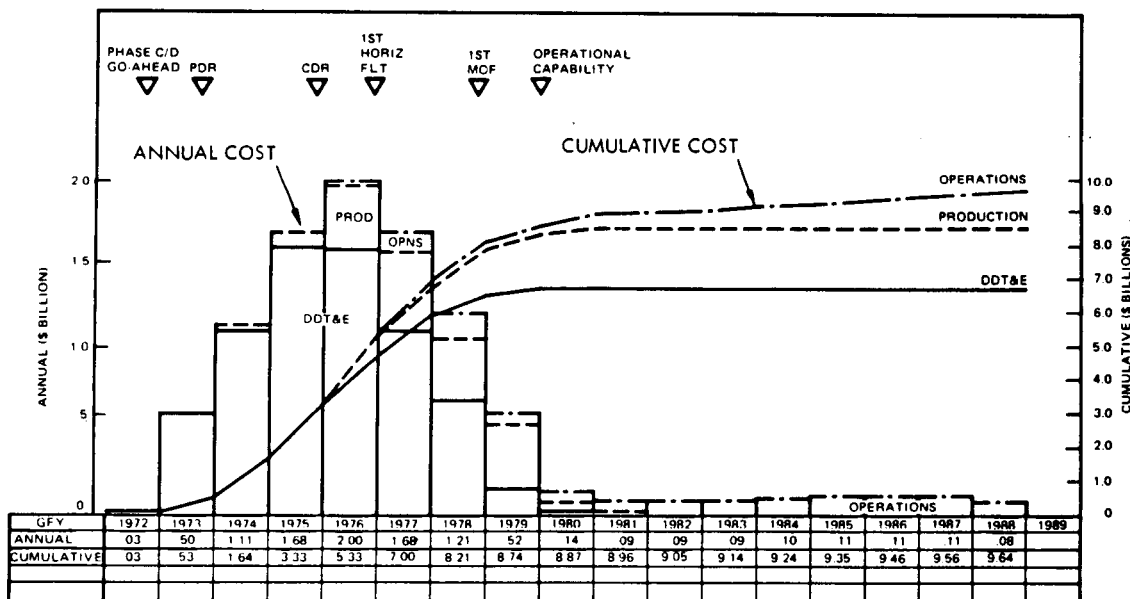Figure 4-59. Space Shuttle Costs, OEHT

Table 4-18. Space Shuttle Costs, OEHT

WBS Level 3 - Description	Cost ($ Million)			
	DDT&E	Production	Operations	Total Program
1.0 Orbiter	$3,394.1	$ 867.4	$ 323.7	$4,585.2
1.1.7 External LH_2 tanks	75.2	518.0	-	593.2
3.0 Booster	2,794.2	355.0	129.0	3,278.2
4.0 Flight test	287.8	-	-	287.8
5.0 Operations	-	-	669.5	669.5
6.0 Shuttle management and integration	157.2	41.8	26.9	225.9
Total costs	6,708.5	1,782.2	1,149.1	9,639.8

TFU orbiter $201,500,000
TFU booster $231,000,000

Both annual and cumulative expenditures are displayed in Figure 4-59; data are segregated into DDT&E, production, and operations costs. Tabular information provided at the bottom of the chart reflects both annual and cumulative costs. Significant program milestones are displayed along the top of the chart to relate the completion of major events with the resources required.

Table 4-19 provides a segregation of the annual expenditures by major category of effort (DDT&E, production, and operations) and by major program element (orbiter, booster, flight test, operations, and management).

The constraints imposed by the master program schedule are reflected directly in these expenditure profiles and are based on customer-directed milestones that require the first horizontal flight to occur in June 1976, the first manned orbital flight to take place in April 1978, and the shuttle to be operational in mid-1979. Phase C/D go-ahead was assumed March 1, 1972, with the first three-month period to be used for resizing the orbiter/booster, refining system requirements, definitizing the shuttle management and

Table 4-19. Annual Space Shuttle Expenditures, OEHT

By Major Category of Effort	Total	GFY '72	'73	'74	'75	'76	'77	'78	'79	'80	'81	'82	'83	'84	'85	'86	'87	'88
DDT&E	6708	27	503	1114	1590	1576	1114	590	161	20	13							
Production	1782			1	63	309	358	367	233	70	38	47	54	62	59	57	64	
Operations	1149				3	23	95	147	76	53	74	90	93	97	107	108	105	78
Total	9639	27	503	1115	1656	1908	1567	1104	470	143	125	137	147	159	168	168	164	78

By Major Program Effort	Total	GFY '72	'73	'74	'75	'76	'77	'78	'79	'80	'81	'82	'83	'84	'85	'86	'87	'88
Orbiter	4585	22	305	614	826	932	861	640	190	67	30	13	16	16	17	17	13	6
Booster	3278	5	180	454	754	841	507	357	179	1								
Tanks	593		6	22	27	43	30	31	28	26	37	47	54	62	61	60	59	
Flight test	288		3	9	19	57	133	40	27									
Operations	699							2	19	34	53	73	73	78	87	88	90	72
Management	266		9	16	30	35	36	34	27	15	5	4	4	3	3	3	2	
Total	9639	27	503	1115	1656	1908	1567	1104	470	143	125	137	147	159	168	168	164	78

technical approach to the C/D program, and discussing the proposed plans and their implementation with the customer.

These schedule requirements have resulted in a definition of expenditure requirements characterized by a modest effort in GFY 1972, a very rapid buildup of resources during the next four years (with peak requirements occurring in 1976), and a sharp decline thereafter through 1980. It is recognized that the execution of this schedule plan will impose a most ambitious undertaking in technical, schedule, and resource management by the customer and its contractors and subcontractors.

4.11.2 PROGRAM COST COMPARISON

The OEHT was compared with the shuttle baseline reusable configuration and a resized three-engine reusable shuttle. All three were costed from weight synthesis computer runs using comparable weight-scaling relationships. The costing was done using comparable and, in many instances identical, cost estimating relationships (CER's). The results are presented in Table 4-20. The reusable three-engine configuration had the lowest program cost, but its DDT&E was only $113,000,000 below that of the reusable baseline. The baseline and OEHT had essentially the same program costs ($9,643,000 versus $9,639,000), and the three-engine reusable had about $100,000,000 lower cost. The crossover (OEHT cost exceeds reusable cost) occurs at 360 flights (see Figure 4-60). A comparison of the DDT&E, final unit, and operations costs is presented in Figure 4-61. The OEHT does provide DDT&E cost savings early in the program and, with the tank production spread over the 13-year program, the tank production costs would not increase the yearly peak fund for 1976.

4.11.3 TANK COSTING

The tank CER's were developed from experience with the S-II tank fabrication. Where design and manufacturing complexity differed, appropriate adjustments were made to the CER's. Figure 4-62 presents a summary of the CER elements compared with the appropriate S-II CER's.

4.11.4 DISCOUNT ANALYSIS

A simple discount analysis was performed on the baseline and OEHT yearly expenditure plans. The present value of each configuration program cost was plotted as a function of the discount rate in Figure 4-63.

Table 4-20. Cost Summary - Heat Sink vs. TPS Booster Configuration

| | Cost ($ Millions) | | | | |
	TFU	DDT&E	Production	Operations	Total
Reusable Configurations					
360-day baseline - TPS booster					
Orbiter, test, & operations	204	3892	904	1054	5851
Booster	286	3208	441	143	3792
Total	490	7100	1345	1197	9643
Three-engine - heat sink booster					
Orbiter, test, & operations	227	4081	982	1065	6129
Booster	242	2906	373	133	3411
Total	469	6987	1355	1198	9540
OEHT					
Three-engine - heat sink booster					
Orbiter, test, & operations	202	3839	909	1020	5768
Booster	231	2794	355	129	3278
External tanks	5	75	518	-	593
Total	438	6708	1782	1149	9639
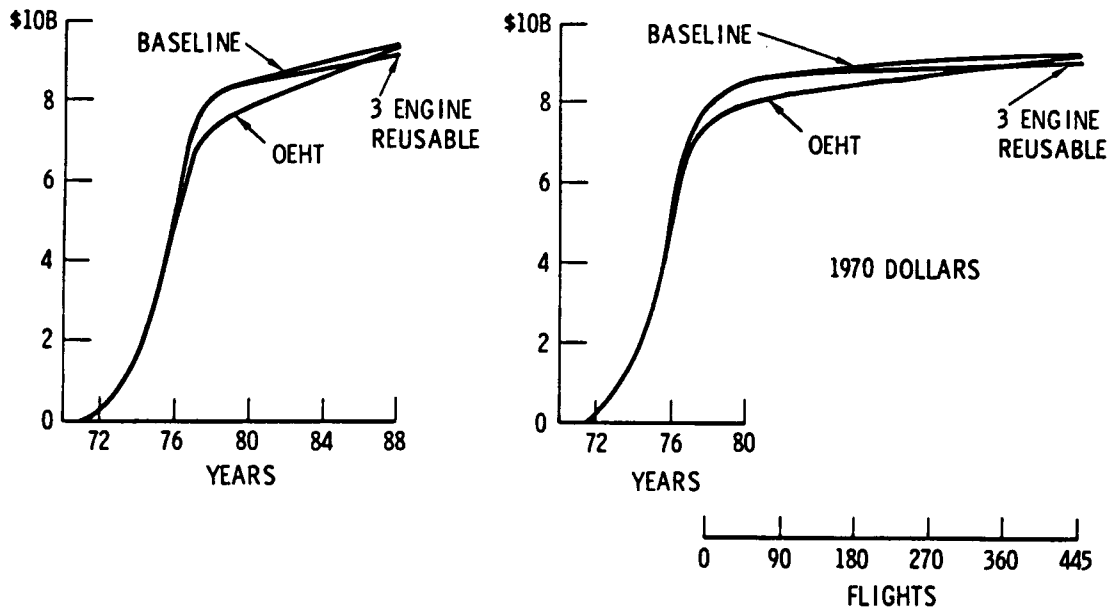
Figure 4-60. Program Cost Comparison
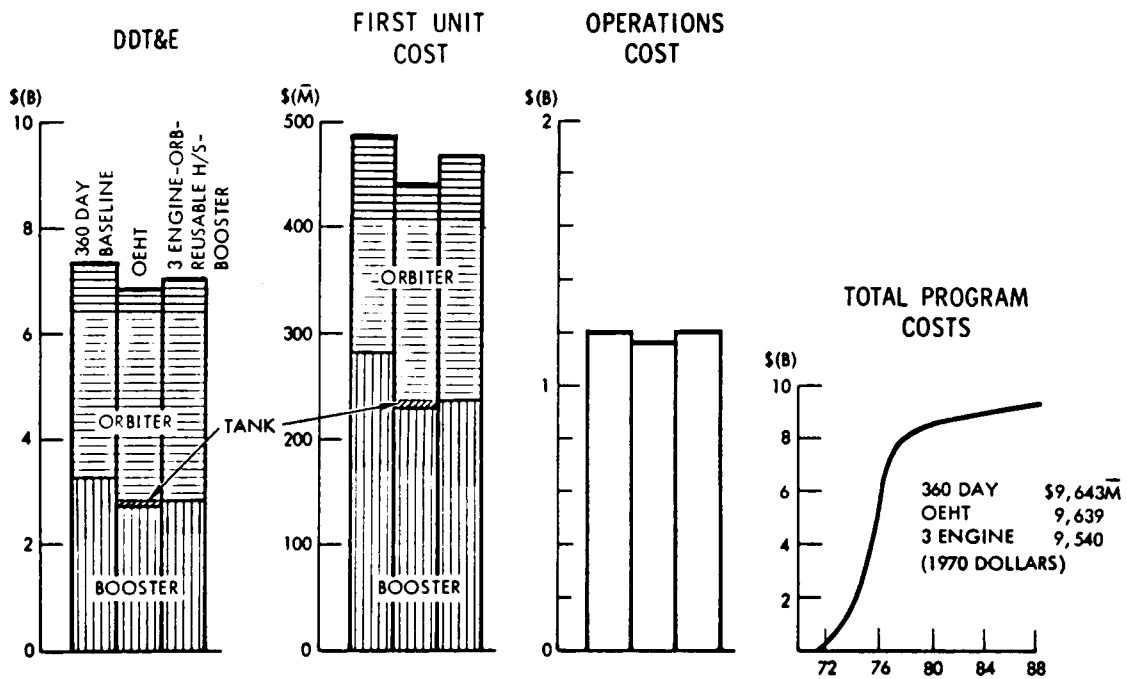
Figure 4-61. Cost Comparison

Figure 4-62. Tank Cost Buildup

DISCOUNT ANALYSIS

2 ENGINE BASELINE
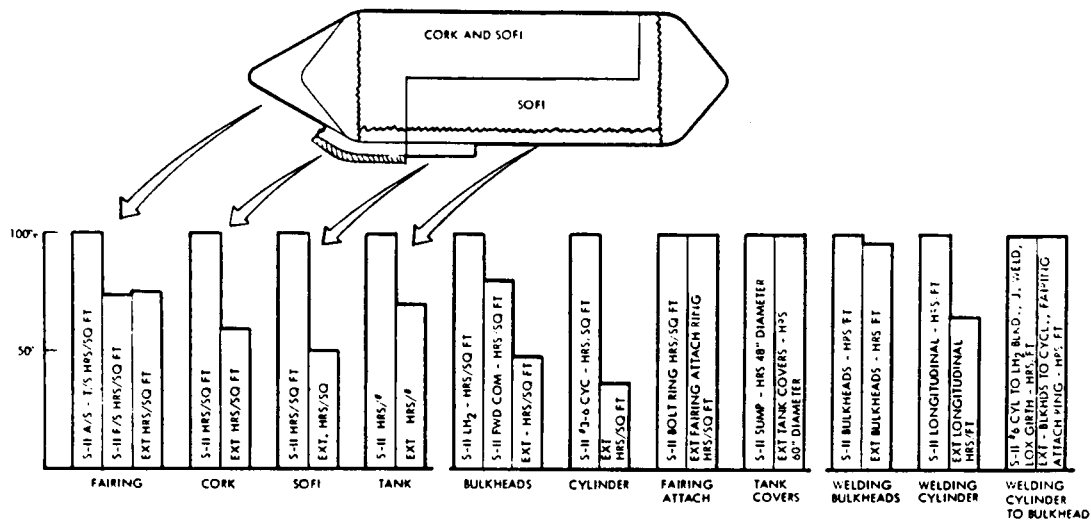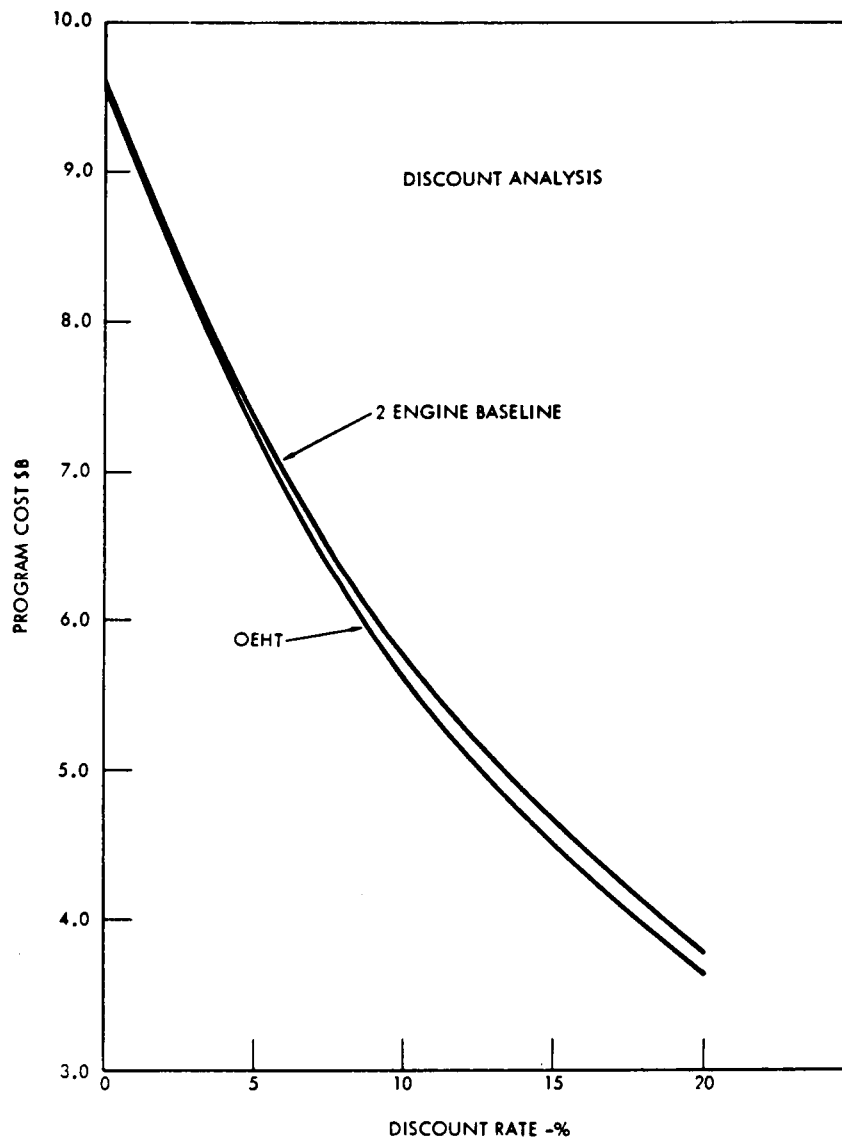
OEHT

Figure 4-63. Discount Analysis

5.0 VEHICLE COMPARISON AND EVALUATION

A summary of the various shuttle concepts developed during the OEHT shuttle study is listed in Table 5-1. The orbiter and booster numbers are included for reference purposes. Of the six shuttle concepts, the following three will be evaluated; all of these use a 730-inch cargo bay.

1. Three-engine external tank orbiter and heat sink booster

2. Two-engine fully reusable orbiter and radiative heat shield booster

3. Three-engine fully reusable orbiter and heat sink booster

The three shuttle concepts to be evaluated were all updated in weight and size to include the latest applicable design, definition, and aerodynamic drag inputs. Complete synthesis weight, performance, and vehicle descriptions were obtained for each of the updated concepts, and layouts were prepared for three orbiters and two boosters.

A comparative vehicle evaluation of the three orbiters is given in Figure 5-1. The external tank 0500 orbiter is the smallest and the fully reusable three-engine orbiter is the largest.

A similar vehicle evaluation of the boosters is presented on Figure 5-2. As noted previously, a booster layout was not prepared for the updated two-engine reusable orbiter with 730-inch payload bay. For this vehicle, therefore, the B9U booster is shown for reference purposes. The B9U booster will not decrease much in size if redesigned. Figure 5-2 indicates the B17-1 booster (for the external tank orbiter) is the smallest and the B9U booster (for the two-engine reusable baseline orbiter) the largest.

The weight comparison of the three shuttle concepts is given in Table 5-2, which are the weights compiled from the individual updated synthesis runs. The external tank orbiter shuttle has the lightest gross lift-off and dry weights; as expected, the two-engine reusable baseline orbiter shuttle is the heaviest system in both lift-off and dry weights.

A final vehicle comparison is shown on Table 5-3, which summarizes system characteristics, weights, costs, and comments on the merit of each system.

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3 ENGINE EXTERNAL TANK 0506 ORBITER	2 ENGINE REUSABLE BASELINE 161C "REV 8" ORBITER	3 ENGINE FULLY REUSABLE 186 ORBITER
• BODY REF LENGTH = 1868 IN. • WING AREA = 6832 FT² • GLOW WEIGHT = 1,027,620 LB	• BODY REF LENGTH = 2108 IN. • WING AREA = 6474 FT² • GLOW WEIGHT = 878,868 LB	• BODY REF LENGTH = 2420 • WING AREA = 7793 FT² • GLOW WEIGHT = 1,365,736 LB
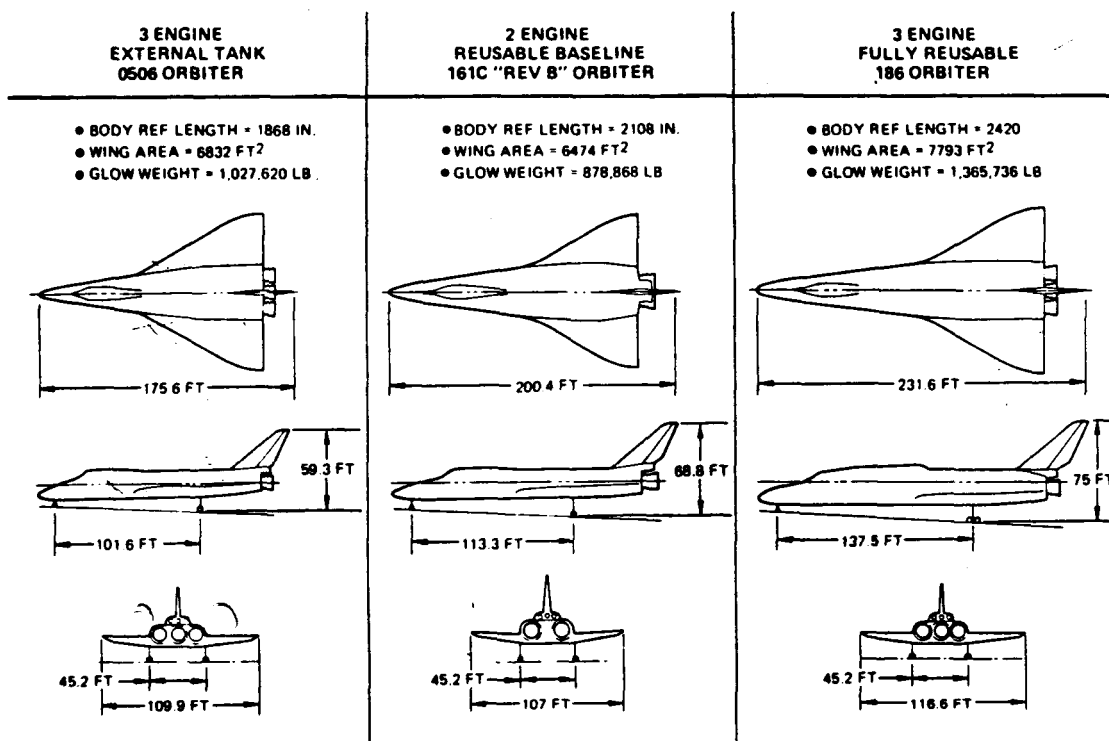
Figure 5-1.   Three Views, 730-Inch Payload Bay Orbiter

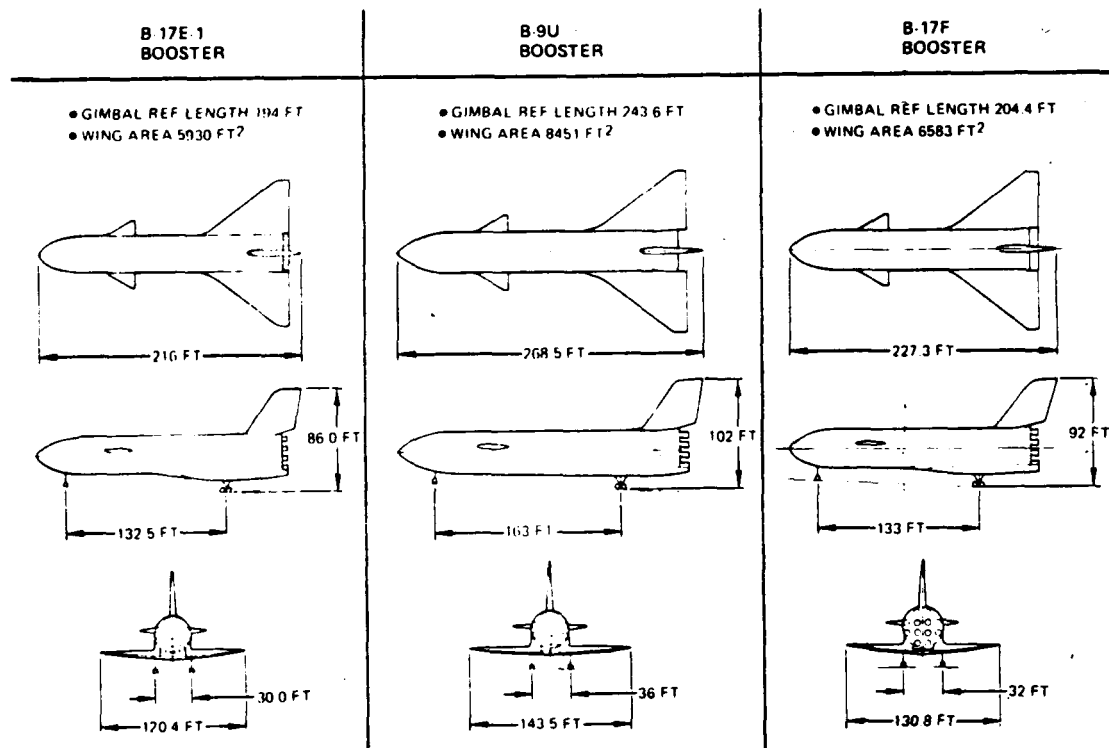| B-17E-1 BOOSTER	B-9U BOOSTER	B-17F BOOSTER
• GIMBAL REF LENGTH 194 FT • WING AREA 5930 FT²	• GIMBAL REF LENGTH 243.6 FT • WING AREA 8451 FT²	• GIMBAL REF LENGTH 204.4 FT • WING AREA 6583 FT²

Figure 5-2.   Three Views, Booster for Shortened
Payload Bay Orbiter

The evaluation of the three shuttle concepts indicates that the external tank orbiter shuttle is the lowest in development costs. Total program costs of the three-engine fully reusable and external tank orbiters are comparable. It is noted, however, that program costs for the external tank orbiter are sensitive factors to be considered and controlled for a successful shuttle program with this concept.

The dispersion of the fragmented tank from an altitude of 350,000 feet is acceptable. Cavity purge requirements for the external tank concept are reduced. In addition, the use of external tanks for the orbiter reduces the test requirement and program risk with respect to consideration of fracture mechanics for the materials of the main LH_2 tanks. The orbiter subsystems and performance characteristics of the three concepts are the same or similar to the all-reusable orbiter, but the heat sink boosters for the three-engine orbiters are simpler to maintain and will have a reduced maintenance requirement.

Table 5-1. Vehicle Comparison and Evaluation

Shuttle		Orbiter	Booster
Designed	Evaluated		
External tank orbiter			
3 engines 800-inch payload bay		0500B	B17E
3 engines 730-inch payload bay Updated	√	0506	B17E-1
Reusable baselines orbiter			
2 engines 800-inch payload bay		161C	B9U
2-engines 730-inch payload bay Updated	√	161C, Rev. B	-
Fully reusable orbiter			
3 engines 800-inch payload bay		185	-
3 engines 730-inch payload bay Updated	√	186	B-17F

Table 5-2. Weight Comparison Shuttle with Shortened Payload Bay Orbiter

Item Weight (lb) / System	Baseline Two-Engine Orbiter	Fully Reusable Three-Engine Orbiter	External Tank Three-Engine Orbiter
Orbiter - dry	224169	292626	216177
External tank - dry	-	-	22117
Booster - dry	607919	480326	439867
Total - dry	832088	772952	678161
Orbiter - GLOW	878868	1365736	1027620
External tank - TLOW	-	-	143610
Booster - BLOW	3861489	3113577	2724840
Total - GLOW	4740357	4479313	3896070
Orbiter - prop	558234	976874	716571
External tank - prop	-	-	119428
Booster - prop	3108606	2549769	2208618
Total - Prop.	3666840	3526643	3044617

Table 5-3. Vehicle Comparison, Shuttle With Shortened Payload Orbiter

Factor	System	External Tank Three-Engine Orbiter	Reusable Baseline Two-Engine Orbiter	Fully Reusable Three-Engine Orbiter
Integrated Vehicle				
GLOW - Polar	(1000 lb)	3,896	4,740	4,479
Payload - Polar	(1000 lb)	40	40	43
Payload - 55 degree inclination	(1000 lb)	25	36	25
Payload - 28.5 degree inclination	(1000 lb)	68	80	76
Vs - Polar	(FPS)	7,719	10,773	7,235
Orbiter				
Main Engines	(lb)	3 @ 477,000 to ???	2 @ 632,000	3 @ 552,000
OLOW *INCLUDES TANKS*	(1000 lb)	1,171 ???	879	1,366
External Tanks				
Concept		No Entry TPS	- -	- -
TLOW	(1000 lb)	144 ???		
Booster				
Concept		Heat Sink	Radiative Heat Shield	Heat Sink
Main Engines	(lb)	13 @ 415,000	12 @ 550,000	12 @ 480,000
BLOW	(1000 lb)	2,725	3,861	3,113
DDT&E Cost	($M)	6,708	7,100	6,987
First Unit Cost	($M)	438	490	469
Tank Cost - Production	($M)	518		
Program Cost - 445 FLTS	($M)	9,639	9,643	9,540
Comments				
Development Costs		Lowest	Highest	Mid
Total Program Costs		Lowest	Highest	Mid
Tank Costs		Sensitive Factor	-	-
Fragmented Tank Dispersion		Acceptable	Mid	Highest
Orbiter Purge Requirements		Reduced	Higher Risk	Higher Risk
Orbiter Fracture Mechanics		Reduced Risk	Same	Same
Orbiter Subsystems		Same	Similar	Similar
Orbiter Aero Performance		Similar	Higher	Higher
Orbiter Tank Inspection		Reduced	Higher	Higher
Booster Maintenance		Reduced	Higher	Reduced

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